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THESIS

AN ANALYSIS OF SPACECRAFT DYNAMIC TESTING AT THE VEHICLE LEVEL

by

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June, 1996

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The US space industry has accumulated a vast amount of expertise in the testing of spacecraft to ensure these vehicles can endure the harsh environments associated with launch and on-orbit operations. Even with this corporate experience, there remains a wide variation in the techniques utilized to test spacecraft during the development and manufacturing process, particularly with regard to spacecraft level dynamics testing. This study investigates the effectiveness of sinusoidal vibration, random vibration, acoustic noise and transient methods of spacecraft dynamic testing. An analysis of test failure and on-orbit performance data for acceptance testing indicates that the acoustic test is the most perceptive workmanship screen at the vehicle level and that additional dynamics tests do not result in an increase in acceptance test effectiveness. For spacecraft qualification, acoustic testing is almost universally employed for qualification in the high frequency environment. For the low frequency environment, data collected from a variety of spacecraft test programs employing sinusoidal sweep, random vibration and transient testing methods shows that a transient base excitation provides the most accurate simulation for the purpose of design verification. Furthermore, data shows that sinusoidal vibration testing provides an unrealistic simulation of the flight environment and results in an increased potential for overtesting.

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AN ANALYSIS OF SPACECRAFT DYNAMIC

TESTING AT THE VEHICLE LEVEL

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ABSTRACT

The US space industry has accumulated a vast amount of expertise in the testing of spacecraft to ensure these vehicles can endure the harsh environments associated with launch and on-orbit operations. Even with this corporate experience, there remains a wide variation in the techniques utilized to test spacecraft during the development and manufacturing process, particularly with regard to spacecraft level dynamics testing. This study investigates the effectiveness of sinusoidal vibration, random vibration, acoustic noise and transient methods of spacecraft dynamic testing. An analysis of test failure and on-orbit performance data for acceptance testing indicates that the acoustic test is the most perceptive workmanship screen at the vehicle level and that additional dynamics tests do not result in an increase in acceptance test effectiveness. For spacecraft qualification, acoustic testing is almost universally employed for qualification in the high frequency environment. For the low frequency environment, data collected from a variety of spacecraft test programs employing sinusoidal sweep, random vibration and transient testing methods shows that a transient base excitation provides the most accurate simulation for the purpose of design verification. Furthermore, data shows that sinusoidal vibration testing provides an unrealistic simulation of the flight environment and results in an increased potential for overtesting.

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I. INTRODUCTION

Over almost four decades of space experience, the United States has successfully developed, tested and launched hundreds of spacecraft. Throughout this period, the US space industry has collected a significant amount of expertise in the most effective techniques for testing space vehicles to ensure they are able to withstand the harsh environments associated with launch and on-orbit operations. Even with this vast amount of corporate experience, there remains a wide variation in the techniques utilized to test spacecraft during the development and manufacturing process. Considerable differences exist among the many government agencies and corporations involved in the US space industry in not only the types and quantities of tests performed but also the severity of those tests. As testing costs can consume up to 35% of a space program's overall budget, the amount of testing performed is of significant impact to overall program cost. Likewise, proper levels of testing are extremely important for program success. If testing is not severe enough, the probability of on-orbit failures is increased; too severe and the space vehicle can be damaged before it is ever launched.

One of the specific areas in which this difference between test approaches is particularly evident is in the realm of spacecraft dynamics testing. Different approaches to dynamic testing of spacecraft at the vehicle level exist not only between NASA and Department Of Defense (DOD) satellite programs but also between the many commercial participants in the US space industry. One example of these differences is the utilization of low frequency sinusoidal vibration testing for spacecraft qualification and acceptance. While many commercial and government space programs include sinusoidal vibration testing as a regular part of the development, qualification and acceptance process, others, including most DOD programs, do not. While the cost savings associated with the deletion of sinusoidal vibration testing can be significant, the potential risks to a program which does not adequately test a spacecraft prior to flight can be considerable.

The purpose of this thesis is to investigate the utility and effectiveness of the various approaches to dynamics testing for both the acceptance and qualification of spacecraft at the system

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level. The spacecraft development and testing process will be described and the various types of

spacecraft dynamic testing will be presented and compared. The underlying fundamentals of

vibration and the spacecraft dynamic environment will also be discussed. A survey will then be

conducted to determine the specific approaches to spacecraft dynamic testing currently in use

throughout the space industry. Test data from a variety of programs will be examined to determine

the effectiveness of these various approaches. Based on this analysis, an approach will be

recommended for dynamics testing of Department of Defense spacecraft.

II. SPACECRAFT DEVELOPMENT AND TESTING PROCESS

Because of the unique nature of space vehicles, the spacecraft development and testing process is often long and complicated. As it is rarely possible to repair problems on orbit, space vehicles must be designed and tested in a robust manner to minimize on-orbit failures. Though the advent of Total Quality Management (TQM) has encouraged the improvement of design and manufacturing processes, a vigorous test and verification program is still necessary to ensure adequate on-orbit spacecraft performance. This test and verification program begins early in the development cycle of a space vehicle and continues throughout the manufacturing, assembly and launch process. While systems level dynamic testing is only one part of this overall development and testing process, it is important to understand the entire process to determine the contribution of dynamics testing to the successful production and launch of a spacecraft.

A. SPACECRAFT DEVELOPMENT AND TEST FLOW

1. Development Phase

The spacecraft development and testing process generally begins with a development phase during which the initial concept is formulated and design work is begun. During the development phase, tests are performed to validate new design concepts, verify analytical models and reduce the risk of transferring a design to actual flight hardware. These tests are often performed on special developmental test articles or breadboard units which are not intended for flight and can therefore be tested under more extreme conditions. Developmental tests are conducted to identify problems early and according to the MIL-STD-1540C, "...confirm structural and performance margins, manufacturability, testability, maintainability, reliability, life expectancy and compatibility with system safety." [Ref. 1: p. 23]

2. Qualification and Protoqualification Phase

Once the initial spacecraft design has been formulated and developmental testing has been accomplished to verify design concepts and reduce the risk of new technologies, the development

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effort generally enters a qualification or protoqualification phase. During this phase, a qualification test article is built. The "qual" vehicle, which is as close to flight quality as possible, is subjected to both functional and environmental tests to ensure that the design, materials and manufacturing processes produce a spacecraft that meets mission specification requirements. The qualification tests certify that both hardware and software work properly and that hardware can survive and operate in the expected environment. Qualification tests also serve to verify analytical models which have been developed to assist in the design verification process. In addition, qualification tests validate the planned acceptance test program, including test techniques, procedures, equipment, instrumentation and software.

To ensure adequate design margins, the qualification vehicle is generally exposed to test levels higher than those expected in flight. Consequently, a spacecraft subjected to qualification testing is generally not intended to be flown and is often utilized as a permanent test fixture. However, to shorten the development cycle and reduce costs, a protoqualification or protoflight approach is often taken. In this case the environmental test levels are lower than those used for the traditional qualification and the qualification article is actually used for flight. When using this approach, refurbishment of the test item may be necessary prior to flight.

3. Acceptance Phase

Once a spacecraft or component is qualified, other articles produced using the same design, materials and manufacturing processes should also meet design specifications and are therefore considered qualified for flight without being subject to qualification tests. However, because materials and manufacturing processes are imperfect, some testing is required to "demonstrate conformance to specification requirements and provide quality control assurance against workmanship or material deficiencies." [Ref. 1: p. 72] Consequently, vehicles or components produced after the qualification vehicle are generally subjected to acceptance tests to demonstrate that the hardware is ready for flight. These acceptance tests are designed to stress items to "precipitate incipient failures due to latent defects in parts, materials and workmanship" [Ref. 1: p. 72] using test levels which are generally not as severe as those used for qualification.

4. Prelaunch Validation Phase

During the prelaunch validation phase, additional testing is accomplished to ensure spacecraft readiness for launch. Such testing can be conducted both before and after shipping to the launch site and is unique for each program depending on transportation methods and the launch vehicle integration process.

Prelaunch operations can include functional testing, Reaction Control System (RCS) loading, pressurization and checkout, ordnance system verification, launch vehicle mechanical and electrical integration and verification, prelaunch countdown simulations and battery charging. Following the successful completion of prelaunch validation activities, the spacecraft is launched and on orbit checkout and operations begin.

B. SPACECRAFT ASSEMBLY LEVELS

The spacecraft development and testing process described above is executed at a variety of levels. Development, qualification and acceptance tests are not only conducted on the entire spacecraft after integration of all components, but each of the components and subsystems which make up the spacecraft are also subjected to tests on an individual basis prior to installation on the vehicle. As the test article becomes larger and more complex, the testing process becomes more costly and difficult to perform and failures are often much more difficult to repair. Consequently, the testing process is often tailored such that testing at the "lower" unit and component levels is more comprehensive and severe than testing at the "higher" system level. This approach is intended to detect and repair failures at the lower levels so that testing at higher levels can be reduced. Because the type and quality of testing done at lower levels can have a significant impact on the effectiveness of tests conducted at the system level, it is important to understand the testing conducted at all levels.

1. Part Level

In actuality, spacecraft testing begins at the part level, where individual parts such as integrated circuits are screened to ensure they are manufactured properly and can withstand the

space environment. If part quality is low, tests at subsequent levels can have an abnormally high failure rate. For DOD space programs, parts are generally required to conform to MIL-STD-1546B, "Parts, Materials, and Processes Control Program for Space and Launch Vehicles," or MIL-STD-1547B, "Electronic Parts, Materials, and Processes for Space and Launch Vehicles." Commercial manufacturers typically invoke similar specifications for parts utilized on their spacecraft. For the purposes of this study, it is assumed that adequate parts screening has been conducted on all programs.

2. Unit Level

The unit level includes components such as electronics boxes, actuators, drive motors and batteries which are "viewed as a complete and separate entity for purposes of manufacturing, maintenance, or record keeping." [Ref. 1: p. 3] Unit level testing is conducted during the developmental, qualification and acceptance phases of unit development and includes both functional and environmental tests such as random vibration and thermal vacuum. Following the successful completion of these tests, a unit is integrated either with a subsystem or directly with the space vehicle for further tests. The type and quality of testing conducted at the unit level can have a significant impact on test results at the subsystem and system level. DOD spacecraft development programs generally conform to the requirements listed in MIL-STD-1540C for testing at the unit level, while commercial manufacturers adhere to similar specifications. The impact of unit level testing on the effectiveness of testing conducted at the system level will be discussed in subsequent sections of this thesis.

3. Subsystem Level

According to MIL-STD-1540C, "a subsystem is an assembly of functionally related units [which] may include interconnection items such as cables or tubing, and the supporting structure to which they are mounted." [Ref. 1: p. 3] While many units are integrated directly with the vehicle and are tested together for the first time at the system level, some units are first integrated and tested together at the subsystem level. Often, subsystems which have special test requirements and which can be tested apart from the entire spacecraft are tested in this manner. A communications

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payload, including associated antennas, is an example of a subsystem which might be tested in this manner. While it is generally considered good practice to conduct tests at the lowest level possible, unit level test requirements are sometimes fulfilled by testing performed at the subsystem level.

4. Vehicle Level

Following the assembly of the spacecraft and integration of all units and subsystems, a series of tests are generally conducted at the vehicle, or system level. These tests include both functional and environmental tests and are intended to ensure that the entire vehicle meets either qualification or acceptance requirements as appropriate for the phase of development. Because many components of the spacecraft are integrated for the first time at the vehicle level, this testing is critical in the detection of problems with such items as wire harnesses, connectors and plumbing which cannot be completely tested at lower levels. In addition, while a high percentage of manufacturing and materials defects are detected at the unit and subsystem level, it has been shown that system level testing still exposes a significant number of unit level defects. [Ref. 2]

The types and order of tests performed at the vehicle level often vary depending on the approach of the manufacturer or agency responsible for development of the spacecraft. As an example, the typical vehicle level test flow for DOD spacecraft as specified in MIL-STD-1540C is shown in Figure 1.1. This test flow specifies the performance of thermal cycle tests first, followed by dynamics tests and then thermal vacuum tests. As a contrast, as shown in Figure 1.2, a vehicle level test flow for the prototype Geostationary Operational Environmental Satellite (GOES) shows dynamics tests followed by limited performance tests and thermal vacuum tests. The flight test flow for a subsequent GOES vehicle thermal vacuum tests followed by dynamics tests. Thermal cycling tests are not performed in either GOES test flow.

C. SPACECRAFT TEST TYPES

A variety of different tests are performed on a spacecraft and its components during the different phases of development. Depending on program requirements, these tests may be performed at the unit, subsystem or vehicle levels, and often may be repeated at all three levels.

These tests include functional tests, thermal and dynamics environment tests, and miscellaneous tests.

1. Functional Tests

Functional tests are intended to verify the mechanical and electrical performance of components at the unit, subsystem and vehicle level. Functional tests are generally performed at ambient temperature and pressure conditions prior to environmental tests to establish a performance baseline. After the test article has been subjected to the required environments, additional functional tests are conducted to determine the impact of the environments on the test article. Functional tests are sometimes executed while a test article is being subjected to an environment, but depending on the environment, this can be complicated and expensive, especially at the vehicle level.

2. Thermal Tests

Thermal environment tests typically include thermal cycling at the unit level, thermal vacuum tests at the unit and vehicle level and thermal balance testing at the vehicle level. Thermal cycling and thermal vacuum tests are intended to verify performance of the test article in the expected temperature and vacuum environments as well as to stress components to detect workmanship and materials deficiencies. Thermal balance testing, on the other hand, is performed to verify that the spacecraft thermal control system is able to maintain the vehicle and its components within required operating temperature ranges when subjected to the thermal environment of space.

3. Dynamics Tests

Dynamics tests can include modal surveys, pyrotechnic shock, random vibration, acoustics and sinusoidal vibration testing. Various combinations of these tests can be performed at the unit, subsystem and system levels depending on the particular test program. Dynamics tests are conducted during the development and qualification phases to verify that the spacecraft structure and components can survive the expected dynamic environment, which is mostly a result of the launch process. These qualification tests also are used to verify finite element models. Dynamics

tests are also conducted during the acceptance phase to stress components to detect workmanship and materials deficiencies.

Figure 1.1. DOD Spacecraft Test Process. From Ref. [3].

Install avionics
Polarity
Continuity
EGSE and SW validity

Install adaptors and appendages
Alignment
Kinematic clearance

Bus functional
Interface
Flight S/W transfer function
Functionality

Install N, S, nadir panels
RF test points
Joint closeout

Install sensor/feed
Calibrate hat/test pt
Calibrate screen
GSE checkout

Payload functional
Check all paths
Performance baseline

Pre-env function
EMC/EMI
CST
Specials (flow/leak)

6.2.1.3 Elec functional
6.2.2 EMC
6.2.6 Pressure

Thermal cycle
Intermediate performance test

6.2.7 Thermal cycle

Dynamics tests
Intermediate performance test
Deployment/ordn

6.2.4 Acoustic
6.2.5 Vibration (1)
6.2.3 Pyroshock
6.2.1.2 Mech functional

Thermal vac
Thermal balance
Performance test

6.2.8 Thermal balance
6.2.9 Thermal vacuum

Post-env test
Repeat pre-env test except EMT/EMC

6.2.1 Functional

Ship/launch ops
Mass properties
Alignments
Preship
Cleaning
Post-ship
Fuel/ordn C-O,
Launch practice

6.2 Special tests

Notes: (1) Random vibration for vehicles under 180 lbs. (400 kg.).

Figure 1.2. GOES Spacecraft Test Process. After Ref. [4].

Static testing also verifies the ability of the spacecraft structure to withstand the launch environment. However, static testing is generally intended to verify only that the structure is able to withstand the quasi-static forces of launch and does not provide verification for the dynamics environment. In addition, the modal survey, which is generally performed on a structural test model or qualification model vehicle to verify structural analytical models could also be considered a dynamic test. These tests will all be described in detail in subsequent sections.

4. Miscellaneous Tests

Additional tests which do not fit into the above categories include inspection, pressure and leakage tests and electromagnetic compatibility (EMC) tests. These tests can be conducted throughout the various phases of spacecraft development and at various levels of assembly. While these tests can be important to the success of a spacecraft development program, they have little impact with regard to dynamics testing and will not be considered in this study.

III. FUNDAMENTALS OF VIBRATION

The term vibration basically refers to oscillation in a mechanical system. This oscillation can be defined by a frequency or frequencies and amplitude. The amplitude may be specified in terms of displacement, velocity or acceleration. Sources of vibration can be deterministic, following a specific pattern such as a sinusoid, or random. In addition, vibration can be free, meaning that no energy is added to the system after an initial disturbance, or forced, meaning that the system is continually disturbed by some input. In order to understand vibration testing, it is important to understand the sources of vibration and their impacts on a mechanical system.

A. RESPONSE OF A SINGLE DEGREE OF FREEDOM SYSTEM

An understanding of the basic phenomena involved in dynamics testing of a spacecraft can be gained by studying a simple single degree of freedom mechanical system. This system, as shown in Figure 3.1, consists of a mass, spring and damper. We wish to define the motion of the system in response to a continuing excitation, or forced vibration. The force may be applied to the mass of the system or to the foundation that supports the system. For a sinusoidal force applied to the mass in Figure 3.1, the differential equation of motion is

$$mx'' + cx' + kx = F_0 \sin(\omega t)$$

The steady state solution for the displacement of the mass can be placed in the form

$$x/x_{static} = R \sin(\omega t - \theta)$$

where

$$x_{static} = F_0/k$$

$$R = 1/[(1-\omega^2/\omega_n^2)^2 + (2\zeta\,\omega/\omega_n)^2]^{1/2}$$

$$\theta = \arctan[(2\,\zeta\,\omega/\omega_n)/(1 - \omega^2/\omega_n^2)]$$

$$\zeta = c/2m\omega_n \text{, and}$$

$$\omega_n = (k/m)^{1/2}$$
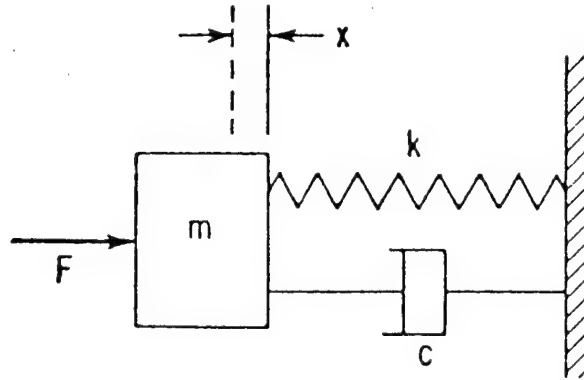
Figure 3.1. Single Degree of Freedom Mechanical System. From Ref. [5].

The dimensionless factor R, commonly called the displacement response factor or magnification factor, provides the relative response of the maximum dynamic displacement to the static displacement that would occur if the force F were applied statically. A plot of the response factor R vs. ω/ω_n and ζ is shown in Figure 3.2. Note that at $\omega/\omega_n = 1$ the magnification factor is limited only by the damping factor ζ. Without damping, the amplitude of the response is theoretically infinite at resonance.

For the purposes of dynamic testing, we are interested in the amount of force transmitted through the system, which can be defined by the ratio of the transmitted force to the applied force. For a single degree of freedom system, the transmissibility is defined by

$$T = R \left[1 + (2 \zeta \, \omega/\omega_n)^2 \right]^{1/2}$$

At resonance, when $\omega/\omega_n = 1$,

$$T_n = R \left[1 + 4 \zeta^2 \right]^{1/2}.$$

Note that for small damping the transmissibility and amplification factor are almost equal. Also, in the forced vibration of a single degree of freedom system, the transmissibility is equal for either mass excitation or base support excitation. Consequently, for a system being excited at its base, such as a spacecraft attached to a launch vehicle, the response is similar.

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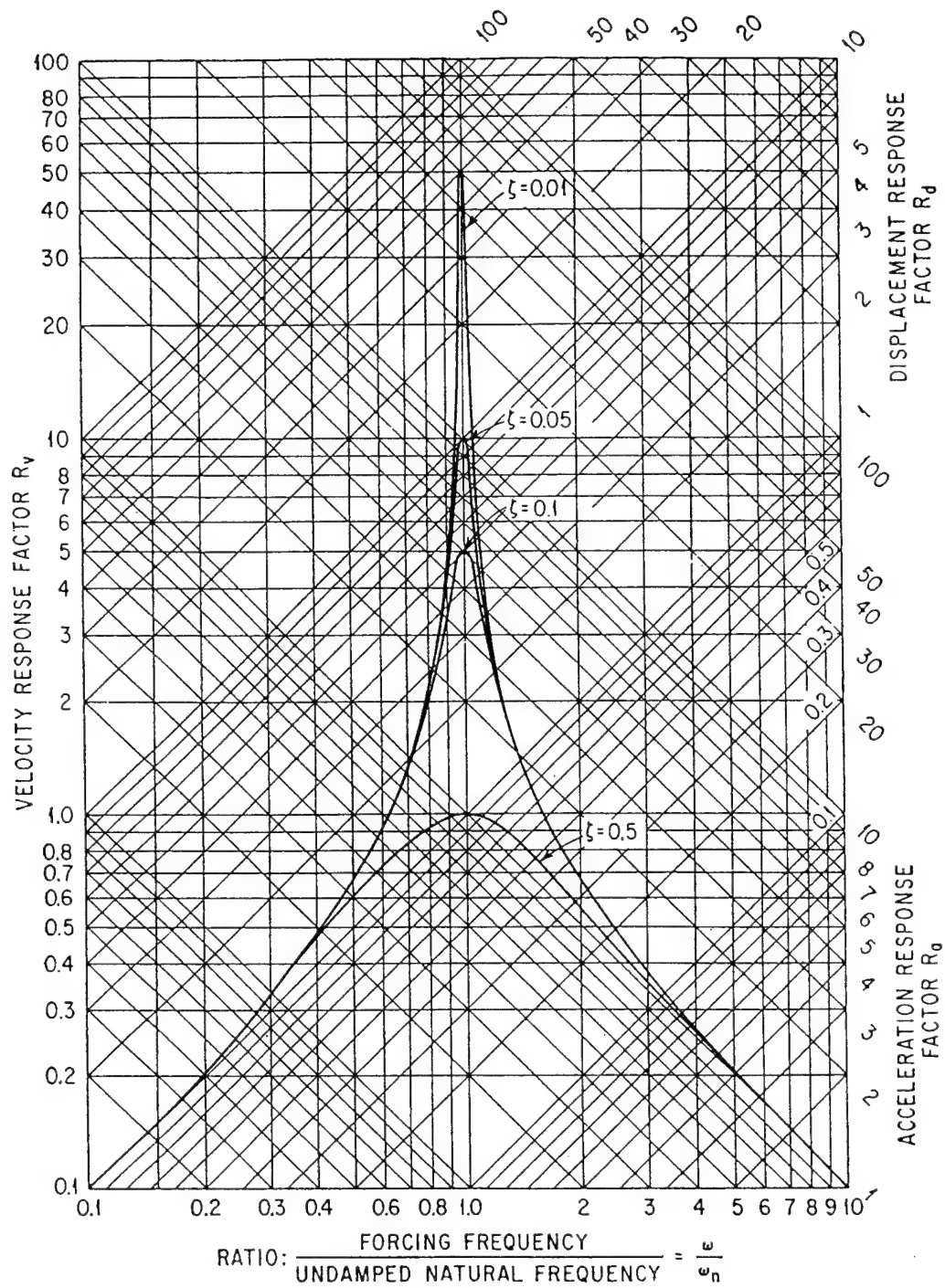
Figure 3.2. Response factor Versus Frequency Ratio. From Ref. [5].

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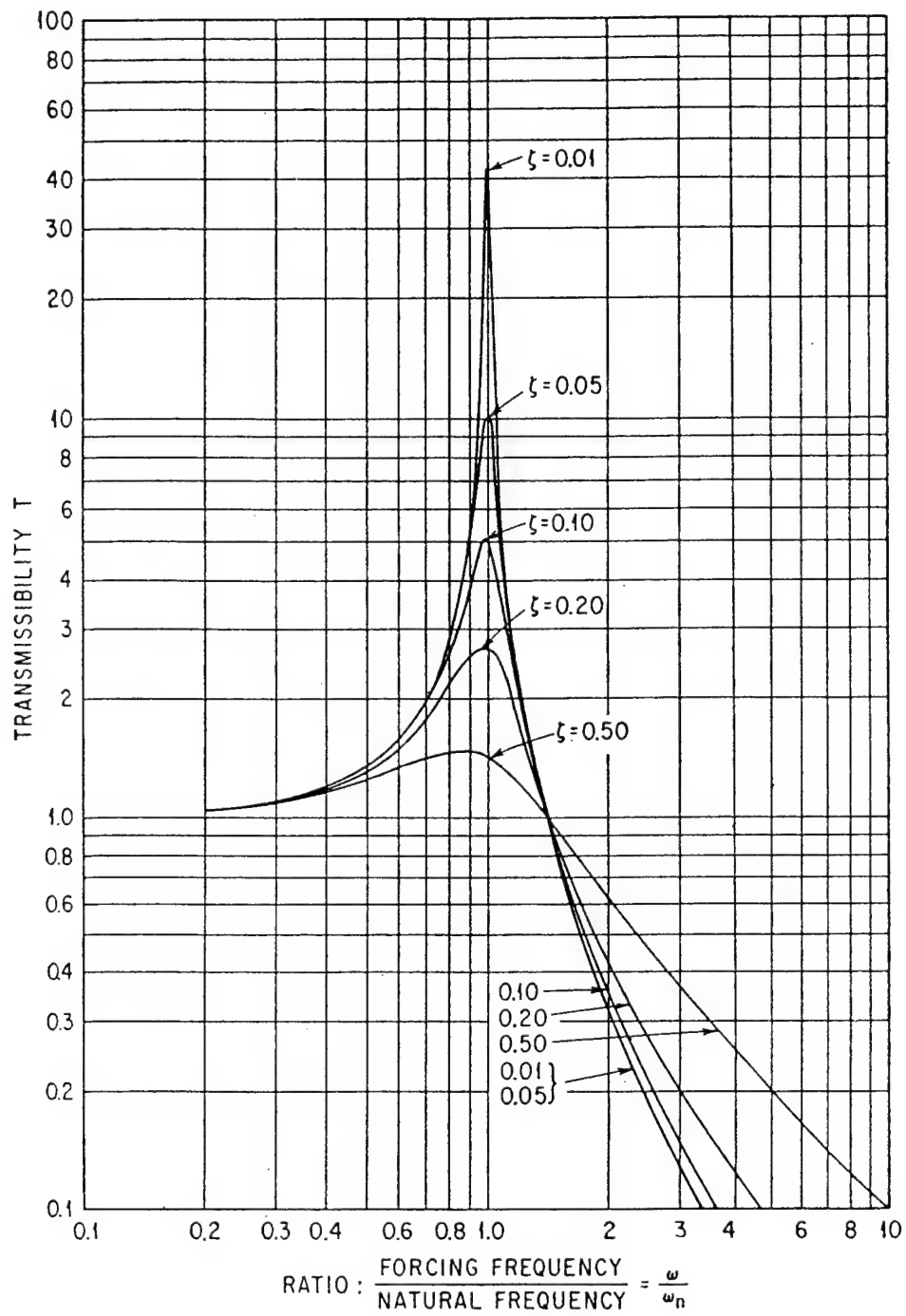
Figure 3.3. Transmissibility Versus Frequency Ratio. From Ref. [5].

Another quantity of interest is the quality factor for mechanical vibration, defined as

$$Q = 1/(2\,\zeta)$$

At resonance, the magnification factor R is equal to the quality factor Q. Therefore, for systems with small damping ratios, the transmissibility T and quality factor Q are approximately equal at resonance. The damping of a system can be estimated from the transmissibility curve at resonance by noting that the bandwidth of the curve at the half power points is $\Delta f = f_n /Q$, as shown in Figure 3.4. Thus, to determine the damping, the transmissibility curve must be obtained by experimental means. Otherwise, the damping must be estimated. [Refs. 5 and 6]
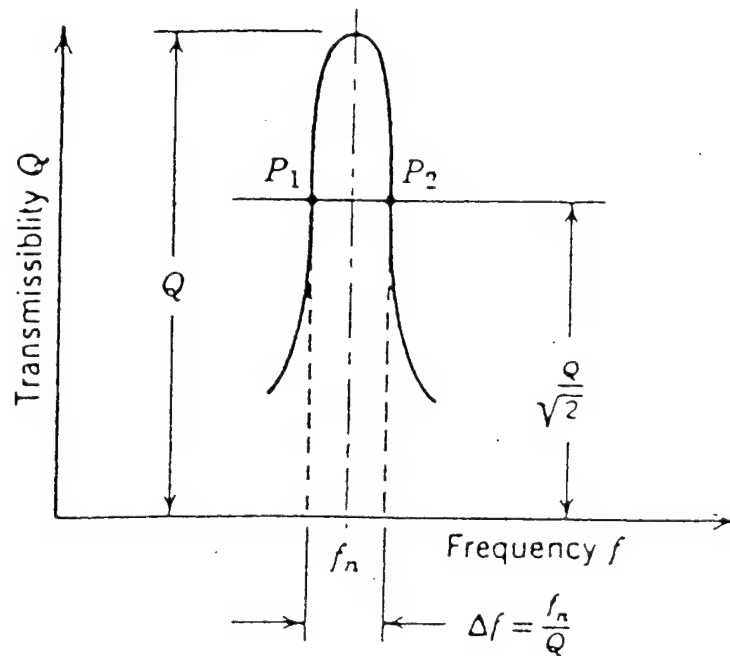
Figure 3.4. Estimation of Quality Factor from Transmissibility Curve. From Ref. [6].

For the purposes of dynamics testing, a swept sinusoidal input is often used as the forcing function. The response spectrum for this swept sinusoidal input can be calculated from an approximate formula. [Ref. 7] Given the steady state response of an excited system as

$$S_O(\omega) = Q * L(\omega)$$

where $L(\omega)$ is the input excitation, the relative response for a swept sinusoidal excitation can be approximated by

$$S_O(\omega) = G * Q * L(\omega)$$

where

$$G = 1 - \exp[-2.86\eta(-0.445)]$$

$$\eta = \beta * Q^2 * \ln 2/(60 * f), \text{ and}$$

$$\beta = \text{the constant sweep rate in octaves/minute}$$

Sweep rates are sometimes stated in terms of a sweep parameter, which includes the terms for damping and natural frequency

$$\text{sweep parameter} = \omega' / (2 \zeta^2 \omega_n^2)$$

where

$$\omega' = \text{the sweep rate in rad/sec}^2$$

A plot of the relative response versus sweep parameter is shown in Figure 3.5. As would be expected, the maximum response decreases as the sweep rate increases.

B. MULTIPLE DEGREE OF FREEDOM SYSTEMS

It is clear that a mechanical system such as a spacecraft is much more complicated than the single degree of freedom system described above. The analysis of these more complex, or multiple degree of freedom systems can be accomplished by representing them as a collection of masses connected by springs as shown in Figure 3.6.

The number of independent parameters required to define the displacement of the masses from their reference positions determines the number of degrees of freedom. For each degree of freedom, a differential equation of motion can be written in the form

$$m_j x_j'' + \sum k_{jk} x_k = F_j \qquad (1)$$

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where

m_j = the lumped mass of the jth degree of freedom

k_{jk} = the stiffness coefficient, and

F_j = the component in x direction of all external forces acting on the mass
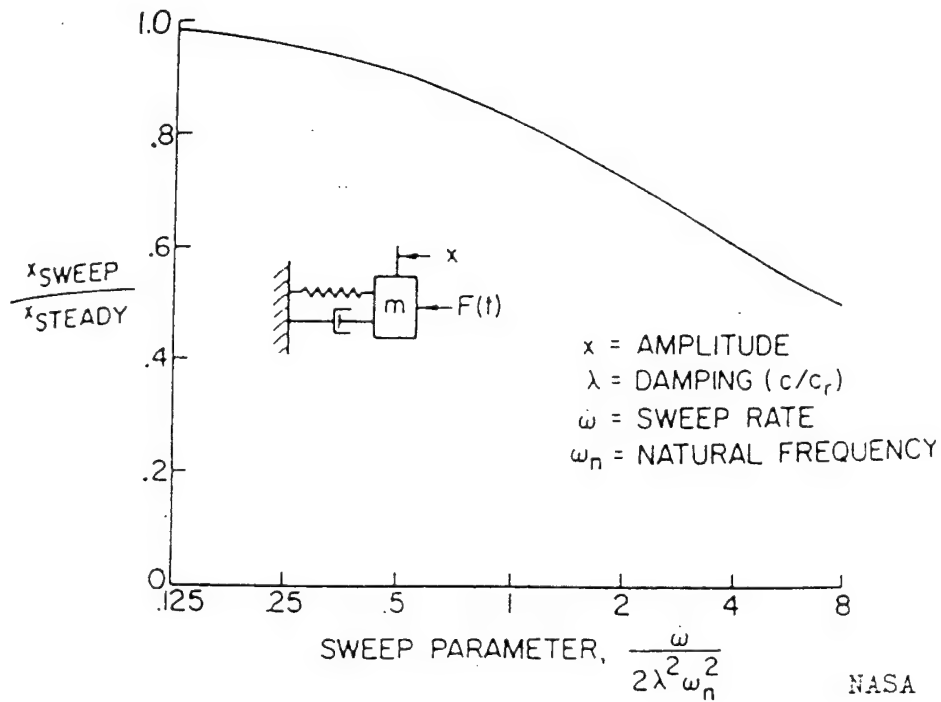
Figure 3.5. Relative Response Versus Sweep Parameter. From Ref. [6].
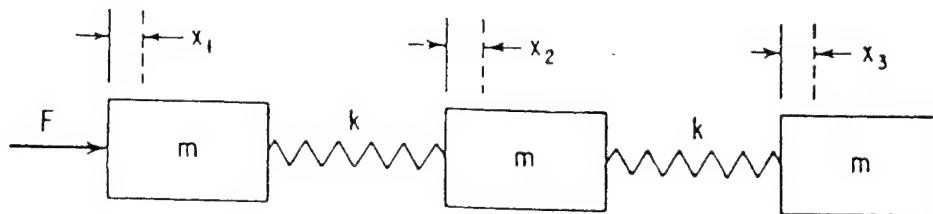
Figure 3.6. Multiple Degree of Freedom Mechanical System. From Ref. [5].

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For $F_j = 0$, this equation has solutions of the form

$$x_j = D_j \sin(\omega t + \theta) \qquad (2)$$

Substituting this solution into the original equation results in

$$m_j \omega^2 D_j = \sum k_{jk} D_k \qquad (3)$$

which is a set of n linear equations with n unknown values of D. A solution of these equations for non-zero values of D can be obtained only if the determinant of the coefficients of D is zero.

$$
\begin{vmatrix}
(m_1 \omega^2 - k_{11}) & -k_{12} & . & -k_{in} \\
-k_{21} & (m_2 \omega^2 - k_{22}) & . & . \\
. & . & . & . \\
. & . & . & . \\
-k_{ni} & . & . & (m_n \omega^2 - k_{nn})
\end{vmatrix} = 0
$$

This solution provides the natural frequencies of the system, ω_n, which are the frequencies at which the system will oscillate in the absence of external forces. For each natural frequency, there is an associated normal mode, or characteristic pattern of amplitude distribution. A normal mode is defined by a set of values of D_{jn} which satisfy equation (3) when $\omega = \omega_n$, or

$$\omega_n^2 m_j D_{jn} = \sum k_{jn} D_{kn} \qquad (4)$$

Another approach to the analysis of more complex mechanical systems is to apply the principle of distributed parameters. A system with distributed parameters has an infinite number of degrees of freedom and can be represented by an infinite number of masses and springs. Since the number of natural frequencies of vibration of a system is equal to the number of degrees of freedom, systems with distributed parameters have an infinite number of natural frequencies. As with the previous analysis, a shape or normal mode is associated with each natural frequency. The complete solution for the free vibration of the system would require the determination of all natural frequencies and modes but in general it is necessary to know only the first few.

For analysis of the forced vibration of an elastic system with distributed parameters such as a beam the classical approach is to apply Newton's second law to derive the equations of motion. For a uniform beam as shown in Figure 3.7

$$EI \, (\partial^4 y/\partial x^4) + \gamma S/g(\partial^2 y/\partial t^2) = F(x,t)$$

where

E = modulus of elasticity

I = moment of inertia of the beam

γ = weight density of the beam

S = area of cross section

Which has the solution

$$y_n = y_{sn} \, [1/(1-\omega^2/\omega_n^2)] \, \sin(\omega t)$$

where

$$y_{sn} = (2Fg/\omega_n^2 S\gamma l) \, (\sin n\pi x/l) \, (\sin n\pi/2)$$

So the amplitude of the nth term of the forced vibration is equal to the static deflection under the Fourier component of the load multiplied by an amplification factor. "This is the same as the relation that exists, for a system having a single degree of freedom, between the static deflection under a load F and the amplitude under a fluctuating load F sin (ωt)." [Ref. 5] As far as each mode is concerned, the beam behaves like a single degree of freedom system. If the beam is
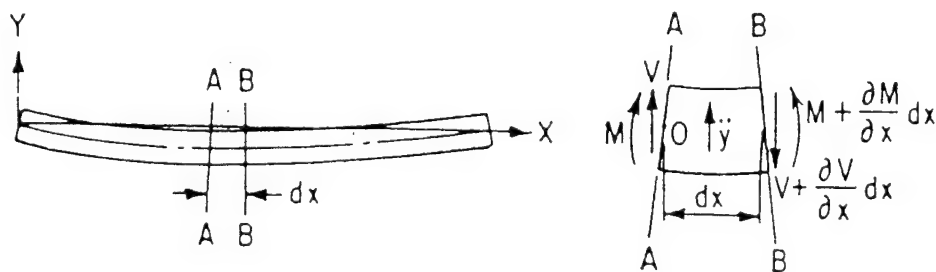
Figure 3.7. Uniform Beam With Distributed Parameters. From Ref. [5].

subject to a force fluctuating at a single frequency, the amplification factor is small except when

the frequency of the forcing function is near the natural frequency of a mode.

> The results for a simply supported beam are typical of those which are obtained for all systems having distributed mass and elasticity. Vibration of such a system at resonance is excited by a force which fluctuates at the natural frequency of a mode. [Ref. 5]

C.   RANDOM AND ACOUSTIC VIBRATION

For the analysis of the single degree of freedom system in Section A, the response was

specified assuming a deterministic, sinusoidal input. Random vibration, however, is non-

deterministic, meaning its instantaneous magnitude is predictable only on a probability basis. Such

vibration may be considered as being composed of a continuous spectrum of frequencies whose

individual amplitudes are varying in a random manner. The amplitude is described statistically by

determining the percentage of time the vibration is within certain limits. Random vibration is

considered to have a Gaussian or normal distribution as shown in Figure 3.8. The gaussian

distribution is described by the function

$$p(a) = 1/[\sigma(2\pi)^{1/2}] \exp(-a^2/2\sigma^2)$$

where σ is defined as the root mean square deviation, or standard deviation of the instantaneous

acceleration from the mean value. For random vibration, the mean is zero so σ is simply the rms
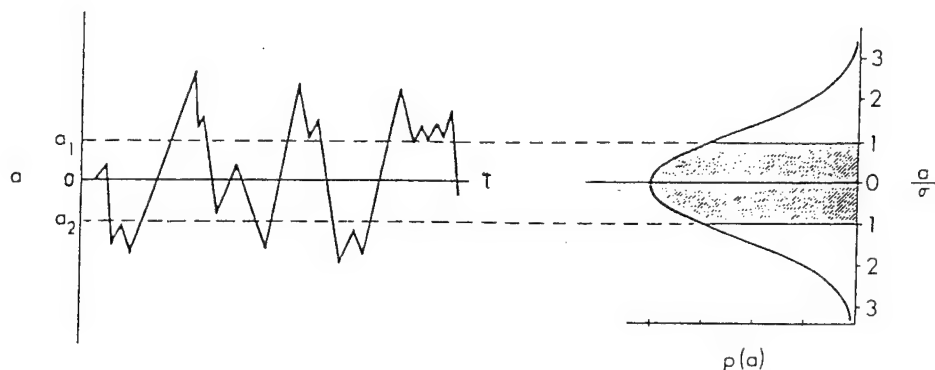
Figure 3.8. Gaussian Distribution of Random Vibration. From Ref. [6].

value of the instantaneous acceleration. For the curve shown in Figure 3.8, the probability that the instantaneous value of the acceleration is between a_1 and a_2 is equal to the shaded area under the curve, or the integral of p(a) from a_1 to a_2.

The vertical scale of Figure 3.8 is in dimensionless multiples of the rms value of the vibration amplitude as defined by σ. The probability, or percentage of time that a given value will lie between the multiples of the rms amplitude (σ) is shown in Table 3.1.

INTERVAL	PERCENTAGE
−σ to +σ	68.27%
−2σ to +2σ	95.45%
−3σ to +3σ	99.73%

Table 3.1. Probability Intervals.

An equivalence between sinusoidal and random vibration can be established by conducting an analysis of maximum damage potential as described in Chapter IV, Section C. For such an analysis, peak values of acceleration are of more interest than instantaneous values. To describe the peak value statistically, we consider a single frequency wave of randomly varying amplitude which could be obtained if a random signal were passed through a narrow bandwidth filter. The envelope of the peak accelerations for this narrow band random sinusoidal wave as shown in Figure 3.9 is described by the Rayleigh distribution

$$p(a_p) = (a_p/\sigma^2) \exp(-a_p^2/2\sigma^2)$$

Since the random vibration contains a continuous distribution of frequencies, all possible structural resonances will be excited and the possibility of damage is much higher than with single frequency sinusoidal vibration.

Random vibration is specified in terms of Power Spectral Density (PSD) in g^2/Hz. As shown in Figure 3.10, a plot of g^2/Hz vs. frequency shows the power distribution or acceleration
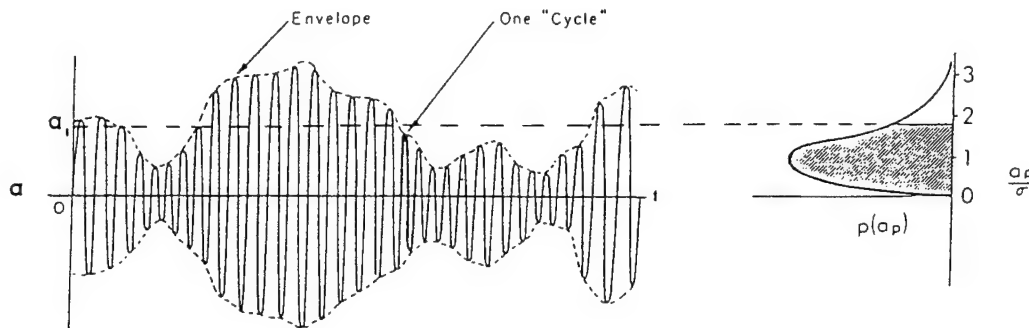
Figure 3.9. Rayleigh Distribution of Peak Accelerations. From Ref. [6].

density of the vibration as a function of frequency. The overall mean square acceleration in g's between frequencies f_1 and f_2 is equal to the square root of the shaded area in the figure, or the square root of the integral of the PSD from f_1 to f_2. Random vibration which has a constant acceleration density is called white noise and the mean square acceleration can be simplified to

$$g_{rms} = (PSD_c * BW)^{1/2}$$

where BW is the frequency bandwidth of interest.
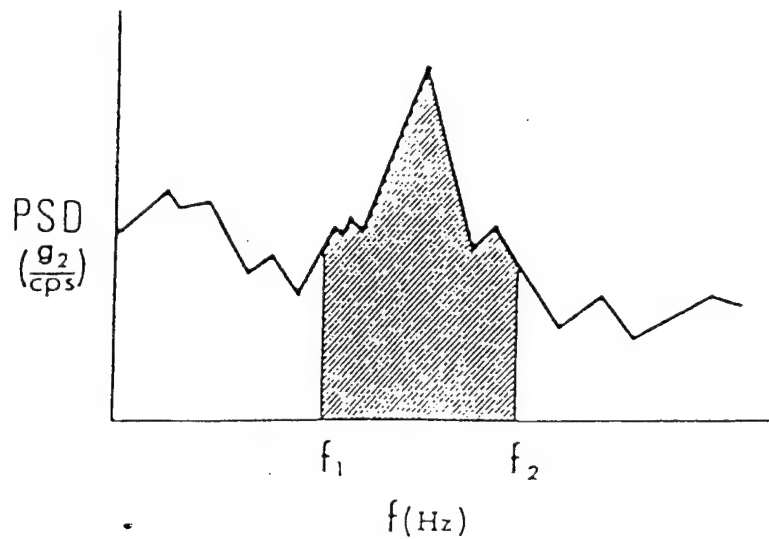
Figure 3.10. Power Spectral Density Versus Frequency. From Ref. [6].

Since all frequencies are present in the random vibration, a system with natural frequencies within the band of interest will be excited at those frequencies by the corresponding single frequency sinusoidal wave. The system responses will be continuous resonant responses with transmissibility described by

$$T = Q = 1/(2\zeta)$$

and the acceleration response at each resonant frequency will be

$$g_{output} = g_{input} * Q \text{ , or}$$
$$g_o^2/Hz = g_i^2/Hz * Q^2 \text{ , or}$$
$$PSD_{out} = PSD_{in} * Q^2$$

So, the PSD of the response at any frequency is equal to the input PSD multiplied by the square of the transmissibility at that frequency.

For a constant or white noise input, the rms response can be calculated as (14):

$$g_{rms} = (\pi/4\zeta * f_n * PSD_n)^{1/2}$$

where PSD_n is the spectral density input to the system at the resonance frequency f_n. Since at resonance $Q = 1/(2\zeta)$, we have

$$g_{rms} = [(\pi/2) * f_n * PSD_n * Q_n]^{1/2}.$$

This expression provides the rms response of a simple system to white noise random vibration at its resonant frequency. It is reasonably accurate for values of Q between 5 and 20 and in practice is used for non-white vibration as well. [Ref. 6]

D.   SHOCK

Mechanical shock is another significant part of the overall dynamics environment which a spacecraft is required to withstand. While the focus of this study is on the comparison of sinusoidal, random and acoustic vibration testing, it is important, to understand the fundamentals of shock analysis and how shock testing fits in with other dynamics tests.

The response of a mechanical system to shock is typically expressed in terms of the shock response spectrum (SRS). In a two dimensional shock response spectrum, the maximum value of

the response in a single time history is plotted as a function of frequency as shown in Figure 3.11. This maximum response can be defined in terms of a displacement, velocity or acceleration. In a three dimensional SRS, shown in Figure 3.12, the distribution of response peaks throughout a time history is displayed as a surface. The two dimensional SRS is most typically employed for shock testing.
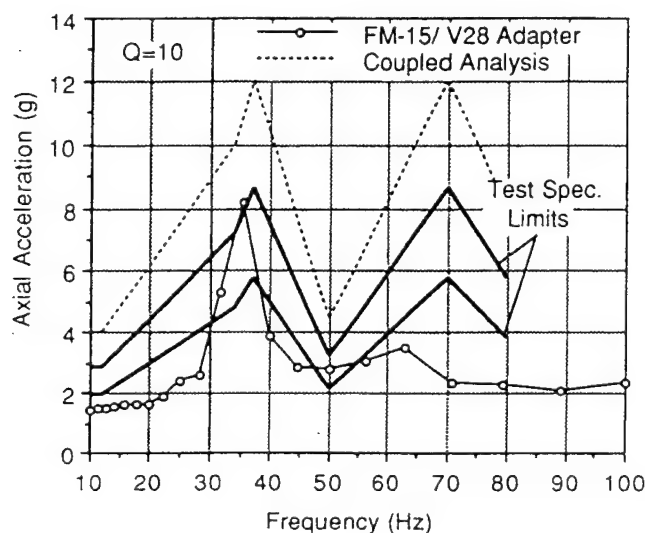
Figure 3.11. Two Dimensional Shock Response Spectrum. From Ref. [8].

Shock response analysis is usually based on the concept of the single degree of freedom system as discussed in Chapter III, Section A. The excitation or input can be an impulse, step, complex or other function in force, acceleration, velocity or displacement as shown in Figure 3.13. The response can be determined analytically using classical differential equations, the Laplace transform or convolution. Time histories and response spectrums for the shock inputs shown in Figure 3.13 are shown in Figure 3.14. Both figures 3.13 and 3.14 are found in Reference 5.
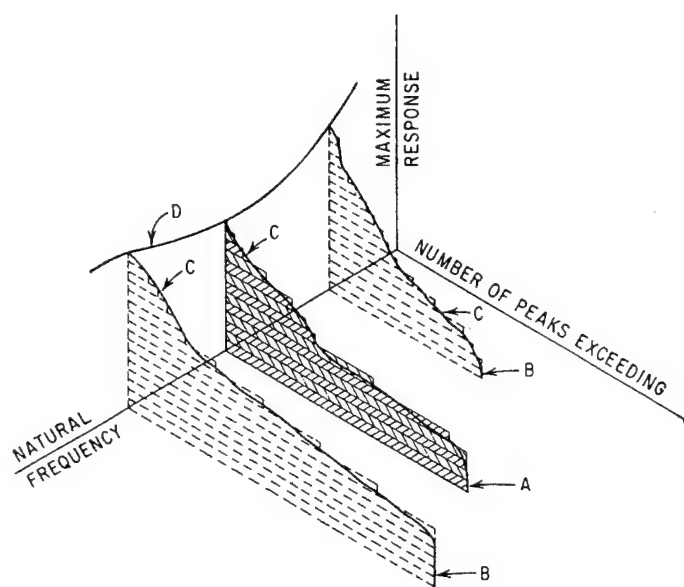
Figure 3.12. Three Dimensional Shock Response Spectrum. From Ref. [5].
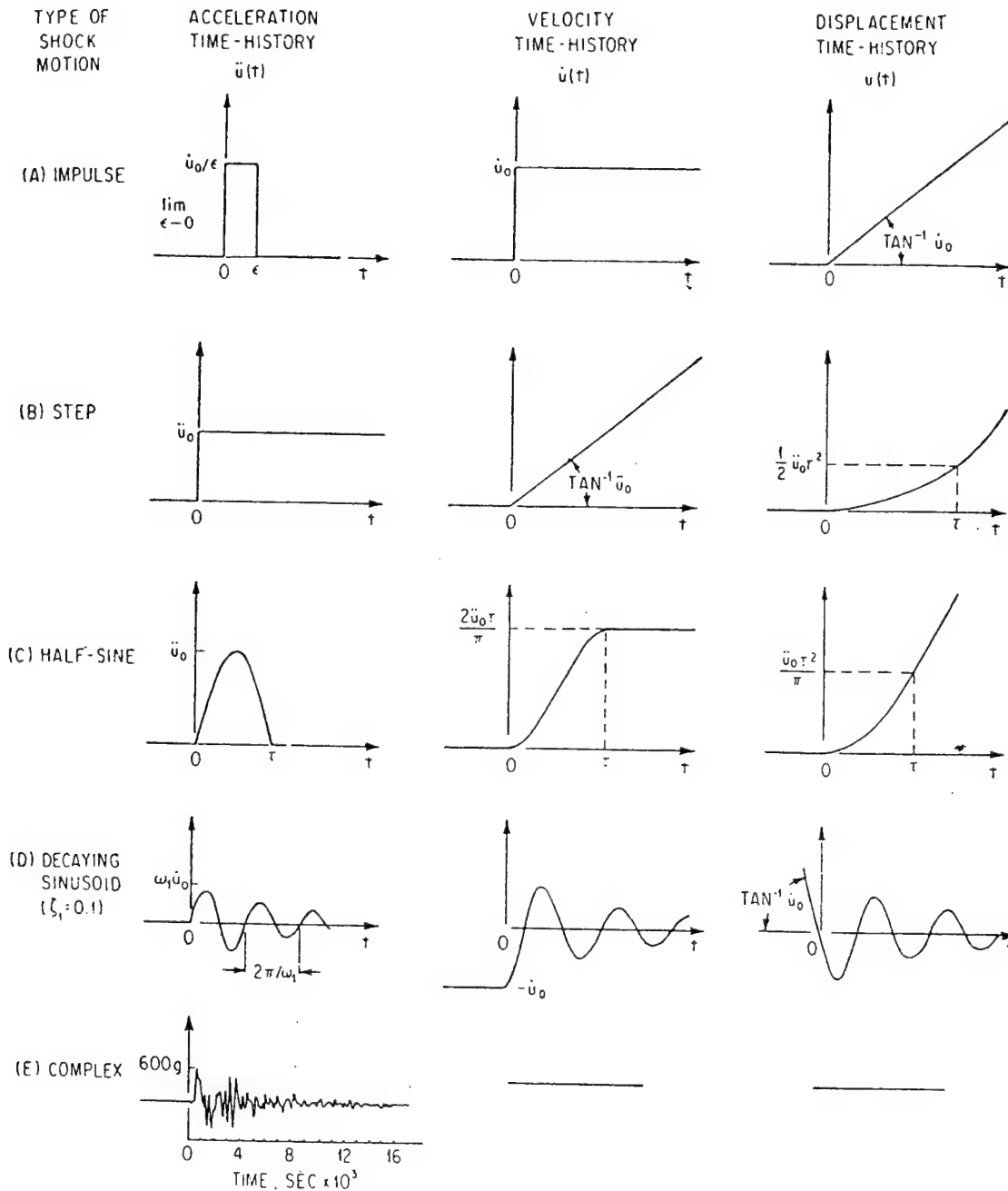
Figure 3.13. Typical Shock Inputs. From Ref. [5].

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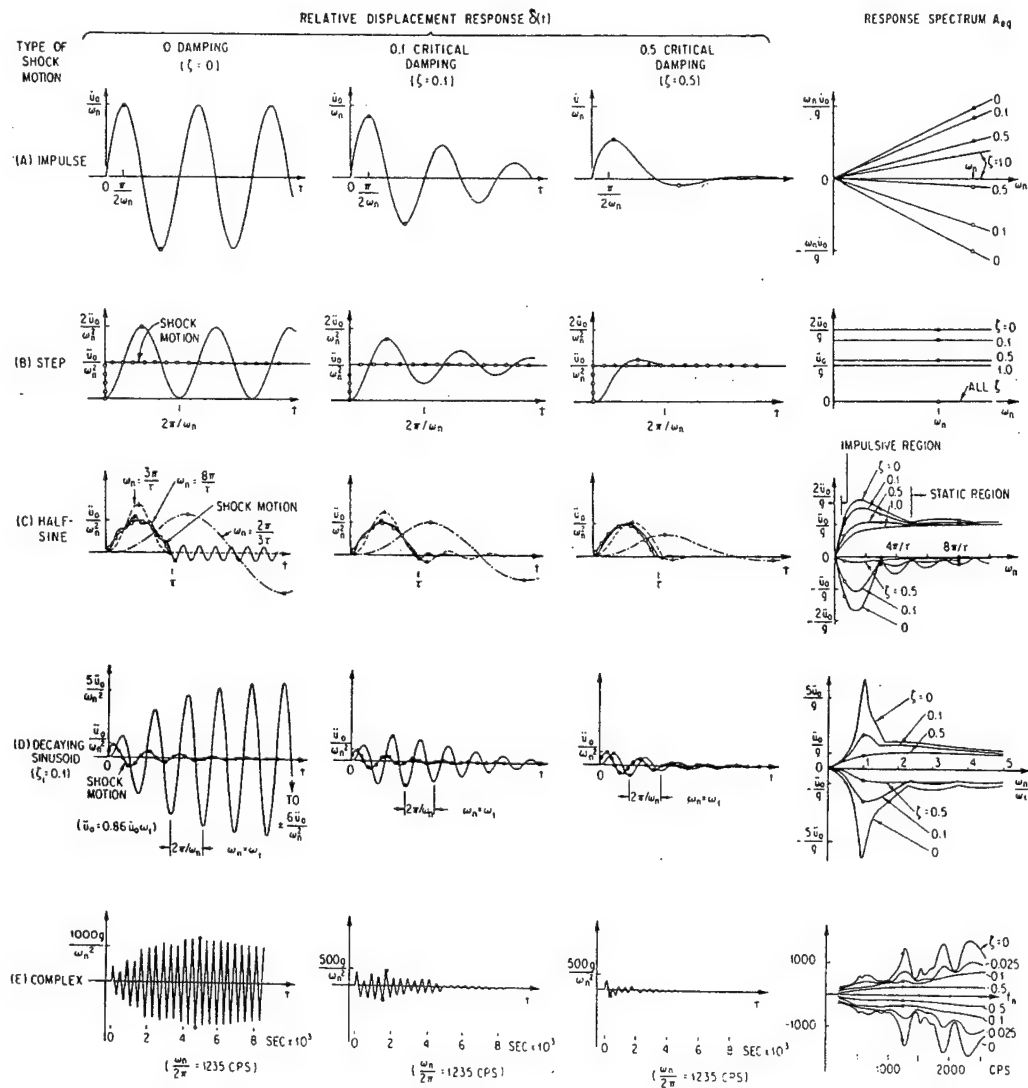
Figure 3.14. Shock Responses for Typical Shock Inputs. From Ref. [5].

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IV. SPACECRAFT VEHICLE LEVEL DYNAMICS TESTING

A. TESTING AND THE LAUNCH ENVIRONMENT

1. Test Intent

Dynamics testing of a spacecraft at the vehicle level is conducted for two primary purposes. The first purpose is that of qualification: to ensure that the hardware can survive and operate in the expected environment. For dynamics testing, this environment consists primarily of launch vehicle induced loads as well as pyroshock from the deployment of mechanisms in flight. Other contributors include ground transportation and handling, but these environments are typically less severe and will not be discussed further.

For qualification, dynamic testing is only one part of the overall process employed to ensure the space vehicle can withstand the dynamic environment. Another important part of the process is the development of an analytical model of the vehicle structure. This analytical model is uded to conduct a simulation to verify that the spacecraft design is adequate to survive the expected dynamic loading. However, even when such a model is developed, testing needs to be performed to verify the accuracy of the model. In addition, testing is also performed to ensure the design can withstand environments which the analysis cannot accurately simulate.

The second purpose of dynamics testing is that of acceptance: to provide quality control assurance against workmanship or material deficiencies. These workmanship or material deficiencies are only problems if they result in the failure of the spacecraft to properly operate in the operational environment. Consequently, for the exposure of material and workmanship problems, the launch environment is again an important factor. Dynamics testing under launch conditions can expose material and workmanship defects that might not be detected in a static condition but which could occur in flight. [Ref. 1]

Two approaches can be used in the application of dynamic tests. The first, environmental simulation, requires the reproduction of the exact mechanical environment to which the vehicle is exposed during launch. While offering realism, this approach can be difficult

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and expensive as it requires a close duplication of the various components of the launch environment. The second, environmental equivalence, requires only that the environment applied during testing has the equivalent effect of the actual launch environment. This approach can be more simple as it allows flexibility in the way tests are conducted as long as the impact is the same. However, it is often difficult to establish either analytically or experimentally whether a test environment is truly equivalent. Equivalent test environments will be discussed in greater detail in a later section.

2. The Launch Environment

The launch loads environment is made up of a combination of steady-state, low frequency transient, higher-frequency vibroacoustic and very high frequency shock loads. A composite sketch of these loads with their frequency and acceleration ranges is shown in Figure 4.1. The overall limit loads can be obtained by combining the root-sum-square (RSS) of the low and high-frequency dynamic components with the steady state component. This results in the specification of the launch limit loads typically published for particular launch vehicles. In addition to the limit load specification, separate specifications for the low frequency transient and vibroacoustic environments are usually provided. These environments are discussed in greater detail in the following sections.

a. Low Frequency Sinusoidal Environment

The primary structural loads experienced by a spacecraft during launch usually occur as a result of quasi-static loads due to acceleration or low frequency launch vehicle bending modes. Launch vehicle modes of vibration which cause significant primary structural loads are generally very low frequency modes, less than 20 Hz. Unless the spacecraft has resonant modes in this frequency range, little dynamic coupling may be expected and the launch loads may be considered as static loads criteria. [Ref. 9] These static loads are usually specified in terms of g limits in the thrust and lateral axis.

For some launch vehicles, specific events such as Main Engine Cutoff (MECO) can result in significant low frequency oscillations at the launch vehicle payload/interface. In these
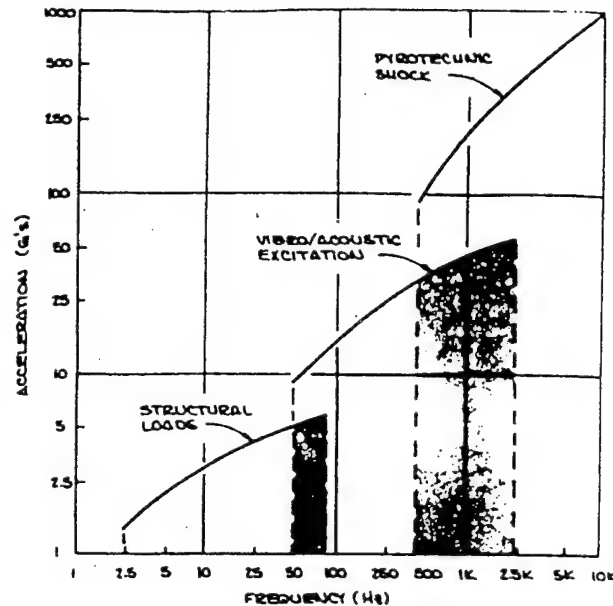
Figure 4.1. Launch Load Spectrum. From Ref. [10].

cases, the low frequency environment is specified in addition to the static loads criteria. Often referred to as the sinusoidal environment, this loading is typically expressed in terms of acceleration amplitude in g over the frequency range of interest, which is usually under 200 Hz.

In addition to specific events such as MECO, low frequency oscillations can be caused by certain launch vehicle phenomena such as POGO and chugging. POGO is a self excited longitudinal vibration which is generated through the closed loop interaction of the launch vehicle structure and propulsion system with combustion chamber pressure and thrust fluctuations. According to Reference 11, POGO is an initially divergent longitudinal vibration which will be stabilized and damped after 10 to 40 cycles. Typical POGO frequencies are 10 to 20 Hz for booster stages, resulting in plus +/- 1 to 2 g output vibration levels at the payload.

During the thrust build up and decay of liquid rocket engines, periodic thrust fluctuations may occur as a result of burning instabilities. Known as chugging, these fluctuations can result in vibration in the range of 60 to 90 Hz for approximately 10 cycles.

b.   *Acoustic Vibration Environment*

The principle sources of the acoustic environment include the interaction of rocket motor exhaust with the surrounding air. This acoustic vibration is a function of rocket thrust, mass flow rate and geometry and generally decays to a negligible level shortly after launch. The second source of acoustic vibration is aerodynamic noise, which is a function of dynamic pressure, mach number and vehicle geometry and is usually greatest in the region of maximum aerodynamic pressure, or max Q. Consequently, the two periods of concern with regard to the acoustic environment are liftoff and max Q.

Acoustic spectra are typically specified in terms of Sound Pressure Level (SPL) in dB versus frequency as shown in Figure 4.2. The spectra are separated into octave band levels, which are frequency bands in which the upper frequency is two times the lower freq. Occasionally the acoustic environment is specified in 1/3 octave bands levels. In addition, an overall acoustic level is generally specified. This level is the sum of the levels over the total frequency band of interest.

$$\text{Overall SPL (dB)} = \sum 10 \log [\, 10^{\,spl(1)} + 10^{\,spl(2)} + ... \, 10^{\,spl(n)} \,]$$

The levels are expressed in terms of dB, defined as

$$dB = 10 \log (P/P_{ref})^2$$

where

P = pressure (rms), and

P_{ref} = reference pressure = average hearing threshold

= 0.0002 μbar = 2.9 x 10^{-9} psi (rms)

Acoustic vibration produces a high energy level over broad frequency range from approximately 20 to 10,000 Hz. Acoustic energy is the primary forcing function causing higher frequency vibrations of flight equipment such as secondary structure and components. In addition, some equipment is sensitive to direct acoustic impingement. This includes items with high ratios of surface area to mass (> 50 in^2/lb) which are exposed to direct acoustic impingement such as solar arrays and antennas. [Ref. 6]
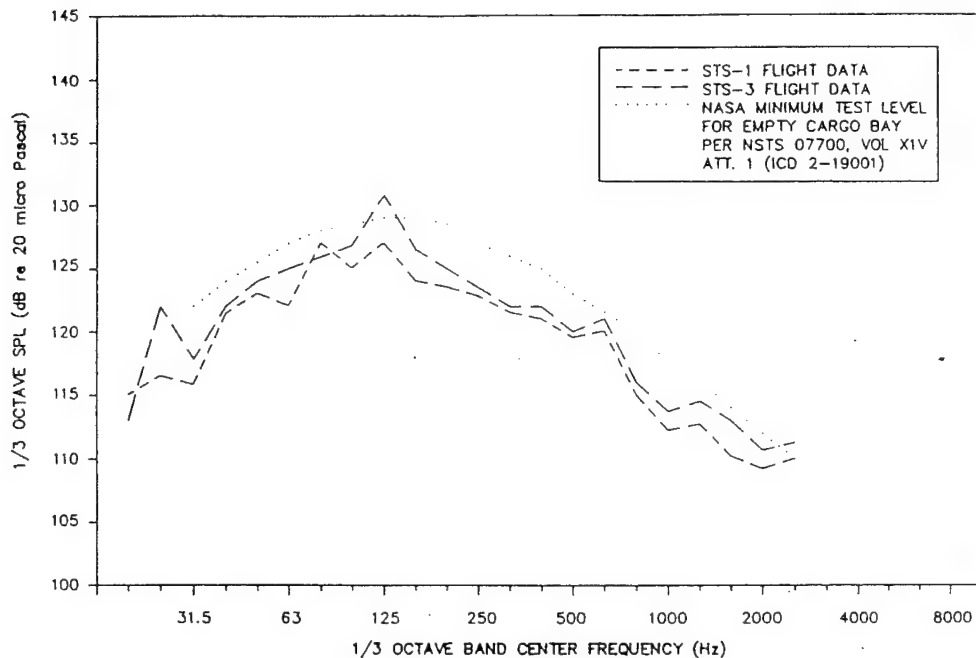
Figure 4.2. Typical Sound Pressure Level (SPL) Plot. From Ref. [12].

The acoustic environment is taken into account throughout the design and test of units and subsystems. The establishment of appropriate vibration test levels at the component level is an important factor in designing for the acoustic environment. Excessively conservative levels can increase the probability of failure while excessively low levels can leave defects undetected. Typical failures in electronic equipment include loose components, detached solder connections, broken leads, cracked connectors and boards and damaged relays. Significant numbers of these types of unit failures are often found in system level acoustic tests even though unit level tests reveal most of the failures.

c.   *Random Vibration Environment*

Random vibration in the launch environment is primarily produced through structural transmission of acoustic pressures impinging on the surfaces of the vehicle. Other, less

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significant sources include structure borne vibration transmitted directly from rocket motors and other equipment. Because the random vibration environment is primarily a product of acoustic noise, acoustic testing is often conducted instead of random vibration at the vehicle level. However, as discussed in more detail in a later section, acoustic vibration does not excite the structure as significantly at low frequencies.

Random vibration is usually quantified in terms of Power Spectral Density (PSD) in g^2/Hz. The PSD is equal to the mean square acceleration (g rms)^2 level in a 1 Hz wide frequency band versus the center frequency of that band. A typical plot of PSD versus frequency is shown in Figure 4.3. The overall rms acceleration in g's may be determined from the PSD by integrating the PSD over the desired frequency range

$$g \ (\text{rms}) = [\int PSDdf \ ]^{1/2}$$

"Random vibration environments are generally not design drivers for primary structure." [Ref. 6] Because the higher frequency (>50 Hz) environments are usually acoustically induced, the random environment mainly affects secondary structure and electronic and electromechanical equipment.
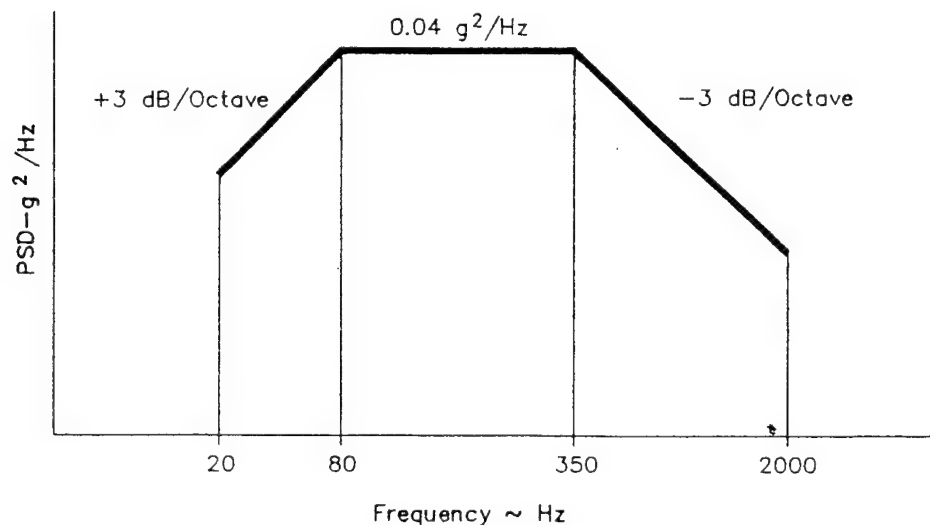
Figure 4.3. Typical Power Spectral Density Plot. From Ref. [12].

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d.   *Shock Environment*

Mechanical shock is generated by a variety of events throughout the launch and deployment of a spacecraft. These events include the separation of booster stages, the separation of the payload from the booster and the release and deployment of stowables such as solar arrays and antennas. In many of these events, pyrotechnic devices such as explosive bolts and nuts and bolt cutters are employed. Consequently the shock environment is often referred to as pyrotechnic shock or pyroshock. Typical shock environments range from hundreds to thousands of g's with durations of 10 to 15 milliseconds.

As discussed previously, the shock response spectrum is the standard industry method for displaying the relative severity of a transient event in the frequency domain. As shown in Figure 4.4, the SRS indicates the maximum response of a simple mechanical oscillator when subjected to the acceleration transient as a base input. The shock response spectrums are specified for a particular value of Q, which is the amplification at resonance of a single degree of freedom system

$$Q = \text{response/input} = 1/2\zeta$$

where

$$\zeta = \% \text{ of critical damping}$$

Thus, the percent of critical damping of the particular structure must be estimated or determined experimentally.

Nearly all failures caused by pyroshock occur in electrical and electromechanical units. Problems include the failure of relays and switches, breakage of leads, cracks in solder and particle contamination. In fact, there are several examples of launch vehicle failures due to chatter or transfer of relays. "Seldom will structure be affected, with the exception of small secondary structural items that may be located in close proximity to an intense shock source such as a linear-shaped charge separation device." [Ref. 13]

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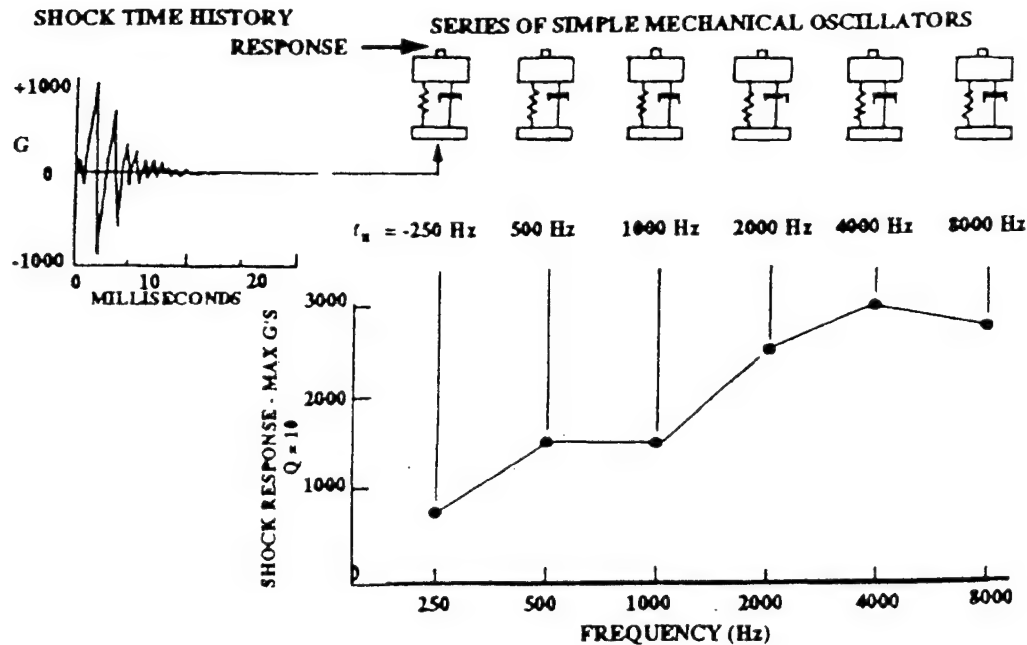
Figure 4.4. Shock Response Spectrum Determination. From Ref. [6].

3. Specific Launch Vehicle Environments

As discussed previously, a primary function of environmental testing is to ensure that spacecraft hardware can survive and operate in the expected environment. Therefore, it is important to have an understanding of the launch environments generated by various launch vehicles. The following section provides a summary of the environments for several launch vehicles and describes how these environments are determined. This information is intended to provide a basic overview and is not intended to be a comprehensive discussion of the environments for the given vehicles.

a. Delta I

The Delta launch vehicle is built in a variety of configurations. These configurations differ in the type of engines used in the first and second stages, the type and number of augmentation (solid rocket) motors and the presence and type of upper stage utilized. The Delta I will be used for an example in this study as there was a significant amount of analysis conducted on the low frequency environment for the NASA Cosmic Background Explorer (COBE) project.

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The Delta launch environment, as specified in Reference 14, includes static load factors, random vibration limit levels, acoustic levels and shock response spectra. The levels for these various environments are provided in Appendix A. In addition, while Reference 13 does not provide specific sinusoidal vibration requirements, it does state that, "There are periodic oscillatory vibrations associated with several events (POGO) during the Delta launch. These occur at frequencies below 50 Hz, and may be amplified by spacecraft dynamics. The need to perform a sinusoidal vibration test to assure survival during these events must be assessed." [Ref. 14] Indeed, for the COBE project, an in-depth analysis was conducted to determine the need for sinusoidal vibration testing and to derive the test levels to be used.

As discussed in Reference 15, sustained periodic oscillation events were found to occur on the Delta I. These events, which manifest themselves as sinusoidal inputs to the payload, are called pre-MECO (Main Engine Cutoff) POGO, and MECO-POGO. Pre MECO POGO, also commonly called mini-POGO occurs at a frequency of 27 Hz and again at 30-40 Hz while MECO-POGO occurs between 15 and 21 Hz. According to Reference 15, the published sinusoidal sweep test levels for the Delta I are generally conservative because they envelope a wide range of spacecraft vehicle combinations. Consequently, the derivation for COBE was accomplished by considering existing flight data from several previous Delta I missions as well as analytic predictions from the COBE/Delta coupled loads analysis for the MECO-POGO and mini-POGO events. "The sinusoidal levels were derived as a COBE unique set of levels. The sweep rate was established so that the time spent sweeping through a particular frequency range during test was approximately the same as the time duration of the event in flight." [Ref. 15: p. 234] The duration of MECO-POGO was about 5 sec which resulted in a 4 octave per minute sweep while the mini-POGO duration was 10 sec which resulted in a sweep rate of 2 oct/min.

The sinusoidal vibration test magnitudes were established using the McDonnell Douglas Astronautics Company (MDAC) coupled loads analysis along with a COBE project analysis of flight data. The flight data included data tapes from 15 missions (14 flights of the 3920 series and one flight of the 3910 series). Shock response spectrum (SRS) plots were developed

from the time histories of launch accelerations obtained from accelerometers on the second stage guidance section. A bandpass filter from 20-40 Hz was employed and SRS plots were generated. A time history for the MECO-POGO event is shown in Figure 4.5 and the corresponding SRS plot is shown Figure 4.6. The SRS plot was developed using a Q of 20, which was based on test data from vibration testing on a COBE Engineering Test Unit (ETU). The SRS was divided by Q to determine the appropriate sinusoidal test level. The flight level was defined as the mean plus 2 sigma and the protoflight test level was then established as 1.25 times flight level.
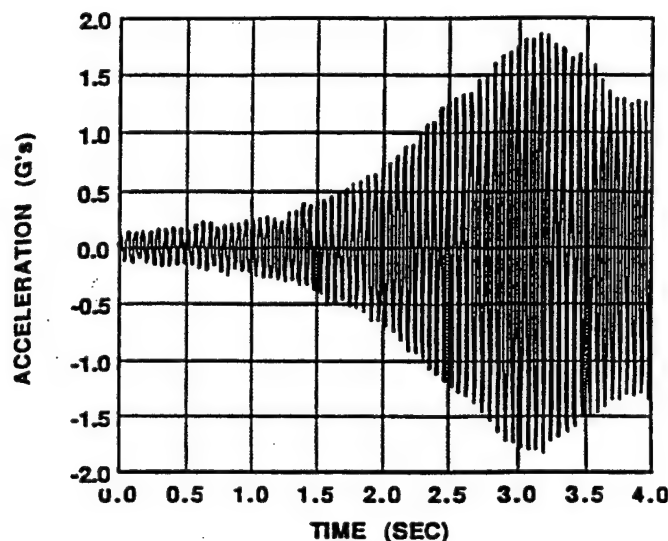
Figure 4.5. Delta 1 MECO-POGO Thrust Axis Time History. From Ref. [15].

The analysis of the flight data showed that the first mini-POGO occurred on all 15 missions. Test levels for this event were set at .15 g's in the thrust direction and 0.50 g's in the lateral direction. For the lateral direction, the level was determined as the root sum square of the pitch and yaw levels. Conversely, only 5 of 15 missions showed the second mini-POGO . A similar analysis resulted in a 0.14 g level in the thrust direction and 0.27 g's in the lateral direction at a frequency of 31 to 33 Hz. For the actual COBE launch, all three events occurred but the levels were lower than test levels as shown in Table 4.1.
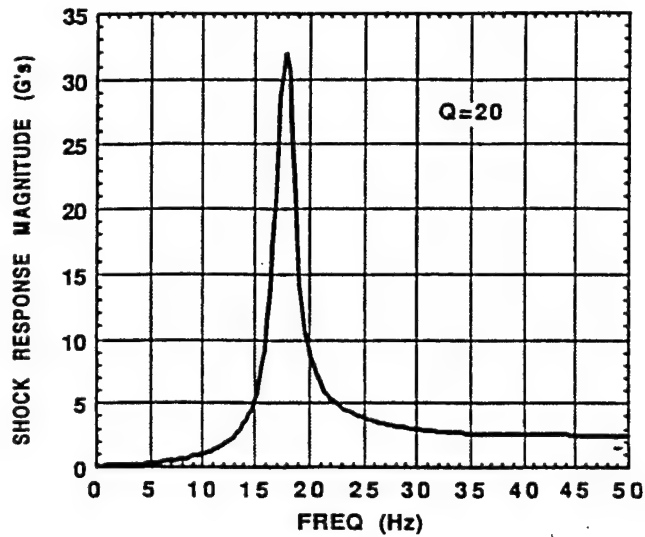
Figure 4.6. Delta 1 MECO-POGO Thrust Axis Shock Response. From Ref. [15].

FLIGHT EVENT/ FREQUENCY	FLIGHT LEVEL (g's)	TEST LEVEL (g's)	TEST/FLIGHT RATIO
Thrust Axis			
MECO-POGO (17.7 Hz)	1.60	2.20	1.38
Mini-POGO I (27.3 Hz)	0.08	0.15	1.88
Mini-POGO II (32.0 Hz)	0.004	0.15	37.50
Lateral Axis			
MECO-POGO (17.7 Hz)	0.35	0.80	2.28
Mini-POGO I (27.3 Hz)	0.14	0.50	3.57
Mini-POGO II (32.0 Hz)	0.05	0.27	5.40

Table 4.1. COBE/Delta I Flight and Test Response Levels.

b. STS

As specified in Reference 14, the environment for the Space Transportation System (STS) or Space Shuttle includes structural loads and acoustic vibration. Structural loads are determined on a case by case basis and acoustic levels are as specified in Appendix A. The mechanical shock environment produced by the shuttle orbiter is considered negligible. In addition, no sinusoidal requirements are specified for shuttle payloads.

c. Atlas

Like the Delta, the Atlas comes in a variety of configurations. Versions of the Atlas currently operational include the Atlas I, II, IIA and IIAS. The Atlas environment includes specifications for limit load factor, acoustic levels, random vibration levels and shock response spectrum. Environments for several versions of the Atlas are provided in Appendix A. While not discussed in Reference 14, the Atlas environment does have a low-frequency quasi-sinusoidal component. As specified in Reference 16, the Atlas II DOD User's Mission Planning Guide, the flight measured low frequency vibration in the 0 to 50 Hz spectrum does not exceed +/-1.0 g axially or +/-0.7 g laterally. These peak responses occur for a few cycles during transient events such as launch, gusts, Booster Engine Cutoff (BECO) and MECO. The Mission Planning Guide recommends that if the spacecraft is tested at the vehicle level with a sinusoidal base vibration the input levels should be tailored to frequency characteristics and response levels consistent with the coupled loads analysis.

B. TEST METHODS

The following section provides an overview of the various dynamic tests employed in the spacecraft development process. These tests can be performed for model verification, for qualification of the vehicle in a particular environment or for workmanship and materials screening. While some tests clearly correspond to a particular environment, others can be utilized to provide equivalence to several environments.

1. Modal Survey

Modal surveys are generally conducted during the development phase to verify that the analytical model of the spacecraft adequately represents the structural dynamics of the actual vehicle. The modal survey determines the natural frequencies and mode shapes of the structure in addition to establishing a lower bound estimate for structural damping. Occasionally modal survey data is used to experimentally derive the structural dynamic model of a vehicle, but this approach is less common. [Ref. 17]

A variety of techniques can be used to conduct modal surveys. These include low level sinusoidal dwell, sinusoidal sweep, random vibration or impact tests. The modal survey can employ single point base or multi-point excitation and typically covers a frequency range from as low as 3 Hz up to 100 Hz. As the goal is to excite the vehicle only enough to determine the natural frequencies and mode shapes, input levels are much lower than those used for qualification or acceptance testing. Once the analytical model is verified using the modal survey results, the spacecraft model can be coupled with the launch vehicle model to obtain a final estimate of flight loads. These flight loads are then used to establish the test levels for qualification and acceptance of spacecraft hardware.

2. Acoustic Vibration

Acoustic tests are often conducted at both the qualification and acceptance levels. The qualification acoustic test demonstrates the ability of the vehicle to operate during and after exposure to the extreme expected acoustic environment in flight. In addition, the qualification test ensures that the level of acoustic testing for acceptance will not result in an over test of the flight vehicle or vehicles. The acceptance acoustic test simulates the flight or minimum workmanship-screen acoustic environment as well as the induced vibration on units in order to expose material and workmanship defects that might not be detected in a static condition. [Ref. 1]

Vehicle level acoustic testing is generally conducted in a reverberation chamber with the vehicle in a stowed configuration simulating conditions inside the payload fairing. As items such as solar arrays and external antennas are especially susceptible to damage from acoustic vibration,

these components are either included in the vehicle level tests or are tested separately at the subsystem level. Acoustic tests typically cover the range from 10 Hz to 10,000 Hz with sound pressure levels from 120 to 145 dB depending on the specific environment and test phase.

3. Random Vibration

Random vibration tests are sometimes conducted at the vehicle level instead of an acoustic test for small, compact vehicles which can be excited more effectively via interface vibration than by an acoustic field. In the case of MIL-STD-1540C, this includes vehicles under 400 lbs. For larger vehicles, random vibration exerted at the launch vehicle/spacecraft interface does not typically excite either primary or secondary structure adequately to simulate the acoustically induced environment. Consequently, the acoustic test is typically specified for these larger vehicles. When conducted, random vibration tests typically cover the frequency range from 20 Hz to 2000 Hz at levels up to 0.05 g^2/Hz.

A narrow band random dwell vibration test has been studied as an equivalent input for the low frequency quasi-static environment. [Ref. 18] Tests were conducted using a 3 Hz bandwidth signal with the input level raised until the critical locations monitored with accelerometers reached the maximum predicted flight response level.

4. Sinusoidal Vibration

a. Fundamentals

Sinusoidal vibration testing of spacecraft at the vehicle level is not consistently employed throughout the space industry. Many manufacturers rely only on structural models for design verification for the low frequency environment, especially below 20 Hz where models are considered fairly accurate. In addition, it is often impractical to vibrate large spacecraft. When it is conducted, sinusoidal vibration testing is usually justified for one or all of the following reasons [Ref. 19]:

1. Sinusoidal vibration provides an equivalent effect for the low frequency launch transient environment which is a significant design driver for primary and secondary structure as well as many assemblies and subsystems.

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2. Sinusoidal vibration is the only environmental test that dynamically excites the lower frequency spacecraft response modes.

3. Sinusoidal vibration provides the only significant dynamics workmanship and design verification for most secondary structure and many assemblies and non-structural system elements.

Sinusoidal vibration testing uses a slowly swept base input sinusoidal excitation to excite system resonances to levels of response which simulate the levels expected to occur during launch. The primary sources of launch excitation considered in developing the sinusoidal vibration test specification are transients produced by events such as engine ignition, burnout and staging. [Ref. 9] Typical sinusoidal vibration levels are in the range of 0.1 to 2.0 g's at sweep rates of 2 to 8 Hz. However, studies have been conducted which employ fast swept sinusoidal inputs of up to 80 oct/min. [Ref. 7]

While the sinusoidal vibration test is considered simple to control and execute, the test is generally very conservative and can result in significant over-test if not executed properly [Ref. 20] For this reason many spacecraft development programs do not conduct sinusoidal vibration testing. To reduce the potential for over-testing, current practice among programs which conduct sinusoidal vibration testing is to employ a technique called notching.

b. Notching

As discussed previously, the low frequency transient environment is typically defined in terms of the shock response spectrum (SRS). Because shock spectrum analysis does not take into account the mechanical impedance of the test item, the characteristics of the spacecraft do not influence the vibration test specification levels. In addition, the vibration test configuration usually does not simulate the boundary conditions and mechanical impedance which the spacecraft sees when it is attached to the launch vehicle. Finally, when subjecting a spacecraft to vibration in a single axis, cross coupling may occur, producing excessive responses in another axis. The overall result is that certain resonant modes of the spacecraft, usually the fundamental bending modes, may be excited to levels of response during the vibration test which exceed the levels

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actually observed during launch. This problem can be solved either by over designing the spacecraft structure to handle these loads or to reduce the test level. Notching has become an accepted method of achieving this reduction. [Ref 9]

Determination of appropriate sinusoidal vibration levels and associated notching levels begins with the mathematical analysis of the vehicle. The mathematical model of the spacecraft structure is coupled to a math model of the launch vehicle to perform launch loads analysis. A complete launch loads analysis includes excitation of the launch vehicle math model with a series of transients which represent each of the events known to result in primary structural loads. A vibration test loads analysis is also done to determine response of the spacecraft structure to the vibration test specification input. Since the sinusoidal input is swept very slowly, the response of the spacecraft at its resonance frequency approaches steady-state conditions. The response at resonance is dependent on the quality factor Q, which is a function of modal damping as discussed previously. Therefore, assumptions regarding modal damping are critical in calculating the response of each mode at resonance. [Ref. 9]

Following generation of the loads analyses, a comparison is made of the vibration test loads with the launch loads to determine if notching is required. If the test response is greater than that for the launch loads, the test input is reduced at the frequency of concern such that the response matches the expected flight response. A typical notched input and corresponding response is shown in Figure 4.7. Notch parameters are usually verified prior to the actual test by conducting a low level sinusoidal sweep survey. Automatic control of sinusoidal sweep tests can also be accomplished by switching the control parameter between input acceleration and response acceleration when the response reaches the specified limit. [Ref. 9]

5. Shock

The shock test demonstrates the ability of the vehicle to withstand or operate in the induced shock environment. This environment includes events such as solid rocket motor (SRM) and payload fairing (PLF) separation as well as booster staging. In addition, the shock environment includes the deployment of components such as solar panels which are initiated by explosive

ordnance or other devices, along with impacts and suddenly applied or released loads. [Ref. 1]

"Shock environments have caused more flight failures in past aerospace vehicle programs (1960-1977) than vibration or acoustic environments." [Ref 6]

Units and subsystems are typically shock tested on vibration shakers or shock synthesizers to levels of up to 3500 g. At the system level shock testing typically includes the firing of all ordnance such as explosive nuts, bolts and pin pullers. This enables the definition of the shock response spectra at various equipment locations. While system level ordnance firing is a good simulation of the shock environment but does not demonstrate design margin.
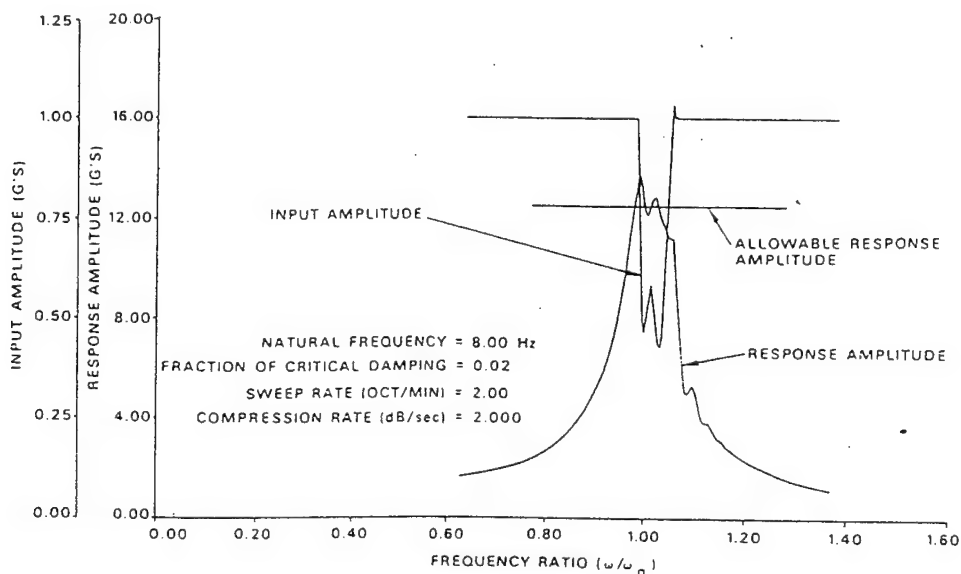
Figure 4.7. Typical Notched Input and Corresponding Response. From Ref. [21].

6.   Transient

Until the advent of digital vibration control systems, low frequency dynamics testing was restricted to swept sinusoidal and random vibration inputs. With the digital control systems, transient waveform control became possible using fast fourier transform algorithms. As will be discussed further in a later section, several studies have shown that a transient waveform test derived from the shock response spectrum can reduce the potential of over test to secondary

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structure. In addition, the transient approach can reduce the number of load cycles to which the test article is exposed.

Transient testing can be conducted to simulate either the shock environment or the low frequency transient and sinusoidal environment. Transient tests are typically conducted by generating a base excitation corresponding to either a standard pulse or to the shock response spectrum (SRS) of the transient of concern. For the input developed using the SRS, the transient waveform with the desired shock spectrum can be synthesized from a pulse train of damped sinusoidal wavelets. The main parameters in shaping the input SRS are the amplification factor Q and the number of sinusoidal wavelets. Amplification factors of 10 (5% damping) to 20 (2.5% damping) are typically used.

Variations in transient testing approaches include [Ref. 22]:

1. Classical pulse--This method uses a classical transient pulse such as a terminal sawtooth or half sinusoidal. Such waveforms are easy to generate on a shaker and excite a broad frequency range, however, they provide a poor simulation of the oscillatory flight transient environments. The classical pulse approach relies on shock spectra to define the magnitude of the input.

2. Modulated sinusoidal pulse--This method was employed in the testing of components in the Galileo spacecraft. While this testing was conducted at the component level, it does have applicability to the vehicle level as well. As described in Reference 22, the Galileo testing employed a series of discrete frequency, limited cycle, modulated sinusoidal wave pulses which were individually applied at the equipment level. At that level, the shape of the input waveform was the acceleration vs. time response of the mass of a single degree of freedom (DOF) system base excited by an exponentially decayed sinusoidal wave transient as shown in the middle of Figure 4.8. For Galileo this was the basic waveform observed for widely separated modes from the Galileo loads analysis responses for a space shuttle launch.

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$$F_1(t) = A \delta(t)$$

$$F_2(t) = D e^{-\zeta \omega_n t} \sin \omega_n t$$

$$F_3(t) = B t e^{-\zeta \omega_n t} \sin \omega_n t$$

$F_1(t)$ = Input Pulse ("Source")

$F_2(t)$ = Pulse at Shuttle/Spacecraft Interface

$F_3(t)$ = Response Pulse at RTG center-of-gravity
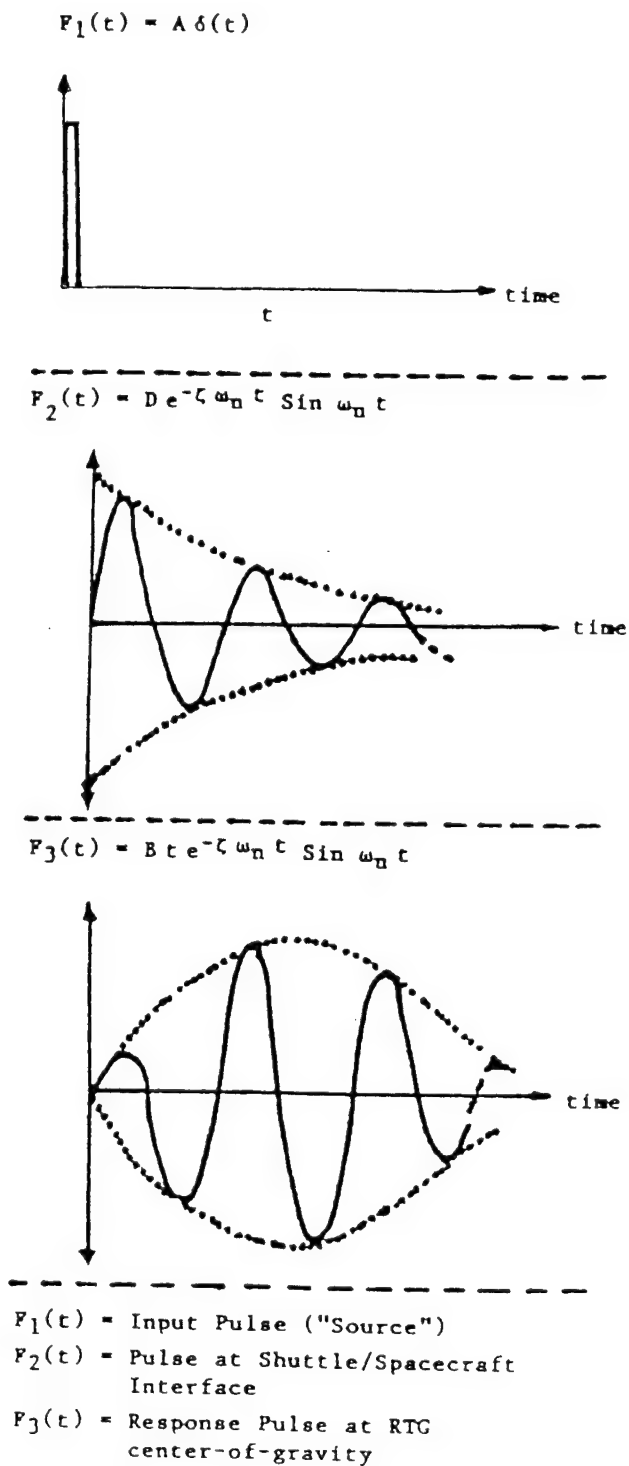
Figure 4.8. Derivation of Galileo Equipment Input and Response Pulses. From Ref. [22].

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Analytically, the transient waveform for the shuttle event of concern was assumed to be a delta function as shown at the top of Figure 4.8. Assuming the shuttle can be represented as a single DOF system, the response at the interface between the Galileo and shuttle was the exponentially decayed sinusoidal wave transient shown in the middle of Figure 4.8. Then, assuming that the spacecraft can also be represented by a single DOF system, the response of the spacecraft at hardware locations was the modulated sinusoidal wave shown at the bottom of the figure. For spacecraft level testing, the input would be similar to that shown in the middle of Figure 4.8.

The principle disadvantage of the modulated sinusoidal pulse is the same as for the swept sinusoidal method. Because the input consists of only one discrete frequency at a time, the test cannot excite all modes simultaneously.

3. Direct transient reproduction--This approach employs a complex transient pulse based on the combined spacecraft and booster loads analysis. While it is the most realistic method, the complex waveform is difficult to generate. In this case, the total system transfer function is estimated and a Fourier transform of the desired response coupled with the transfer function is used to produce a shaker transient in the frequency domain. The time domain transient is then obtained from the inverse transform and is applied to the spacecraft. The resulting and desired responses are then compared and the results used to fine tune the system transfer function. As described in reference 6, this process is repeated several times until a pre-determined convergence criterion has been met.

C. TEST EQUIVALENCE

1. Sinusoidal and Random Vibration Equivalence

While considerable effort has been expended to develop equivalences between sinusoidal and random vibration specifications, a "large number of investigators have concluded that no reliable equivalence technique exists." [Ref. 6] None-the-less, several equivalence techniques

have been developed. These techniques are generally based on one of two approaches. The first is the maximum damage potential to the system, which is associated with the peak load criterion. The second is the cumulative damage to the system, which is associated with cumulative fatigue damage.

The maximum damage potential to a system is assessed by letting the sinusoidal vibration be the reference vibration. The sinusoidal peak response at resonance is then equated to the 3-sigma peak, or other desired estimate of peak, for the random vibration. Other methods reduce the sinusoidal peak response to the rms response and then equate the result to the equation

$$(g_{rms}) = (\pi/2) * (PSD)_n * f_n * Q_n$$

where

$(PSD)_n$ = the spectral density input to the system at the resonant frequency f_n, and

Q_n = the system quality factor at the resonance frequency = $1/2\zeta$.

This relation is often used to find the input PSD spectrum by estimating Q at various frequencies across the spectrum, which is basically the same as estimating damping. "This procedure assumes the system may have resonances at any frequency in the band of interest." [Ref. 6] If the random vibration is the reference, the procedure is reversed.

Fatigue considerations are more complex but are important for systems subjected to random vibration because of the continuous excitation of system resonances due to the presence of all frequencies simultaneously. The equivalence is based on the cumulative damage concept and a synthesized S/N curve for a large class of structural materials. [Ref. 6] The following theoretical predictions have been developed for the number of stress reversed cycles required to produce failure in typical materials used in engineering structures:

$$N_{(sine\ vibe)} = (5 \times 10^6)/(u''/u''_{el})^{6.5}$$

$$N_{(random\ vibe)} = (3.33 \times 10^4)/(x''/x''_{el})^{6.5}$$

where u'' is the sinusoidal acceleration input to the equipment under vibration and x'' is the rms response acceleration in random vibration. The subscript el applies to accelerations corresponding to the endurance limit stress for the material in question. [Ref. 6] Setting the number of stress

reversals to failure equal for sinusoidal and random vibration and applying the equation for g(rms) yields

$$grms(\text{sine input}) = 2.72[(PSD_{in} * f_n)/Q]^{1/2}$$

which provides the sinusoidal acceleration input equivalent to a random vibration PSD input.

2. Test Duration Relationships

It is sometimes useful to compare the equivalent duration of one type of test with another. For a comparison of random and sinusoidal inputs, this equivalency is often performed on the basis of the number of cycles of the particular input. For a relatively short sinusoidal sweep input, we are concerned only with the number of cycles of the sinusoidal forcing function which are at or near the resonance frequency of the structure under test. Given the frequency at time t for a logarithmic sinusoidal sweep rate

$$f(t) = f_o * 2^{\beta t/60}$$

where

f_o = initial frequency in Hz at t = 0, and

β = the constant sweep rate in octaves/minute.

The number of cycles within the half-power bandwidth of the resonance frequency is

$$n_i = (60 * f_n)/Q * \beta * \ln 2$$

where

Q = the generalized amplification factor $(1/2\zeta)$, and

f_n = the resonant frequency of the test item. [Ref. 7]

Correspondingly, the number of cycles applied during a random test at the resonance frequency of the test article is easily determined as

$$n_i = T * f_n$$

where

T = the test duration. [Ref. 11]

The above relationships illustrate one of the most significant differences between the swept sinusoidal and random tests. The swept sinusoidal test provides a significant input at each

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resonant frequency only during the period when the forcing function is sweeping through the half-power bandwidth of that particular resonance. The random signal, however, excites each resonance frequency within the bandwidth of the input simultaneously throughout the entire duration of the test.

V. VIBRATION TESTING ANALYSIS

A. TEST PHILOSOPHIES

Dynamics testing practices vary significantly between the various government agencies and corporations involved in the development and manufacturing of spacecraft. These different practices have been developed over the years for a variety of reasons, including the availability of facilities for testing, cost and practicality of performing certain tests and the actual or perceived effectiveness of tests. The following section provides a summary of the dynamics testing approaches employed at the spacecraft level by a variety of organizations. The differences in these approaches will provide the basis for the study discussed in later sections.

1. Department of Defense

Spacecraft level dynamic test requirements for DOD payloads are specified in MIL-STD-1540C. These requirements are summarized in Table 5.1. At the qualification level, shock tests are required for all vehicles. Acoustic tests are required for all vehicles with mass greater than 180 kg while random vibration tests are required for payloads smaller than 180 kg. Levels for shock, acoustic and random vibration qualification tests are defined as the extreme expected environment, which is the level not exceeded on at least 99 percent of flights, estimated with 90 percent confidence (P99/90). The specification also requires a modal survey to verify the analytical model and define the resonant frequencies, mode shapes and damping for all modes up to at least 50 Hz.

While the MIL-STD-1540C specification requires no qualification testing to simulate the low frequency quasi-static environment, it does discuss the extreme and maximum expected sinusoidal environments. According to the specification, sinusoidal vibration acceleration amplitudes are considered significant if they exceed 0.016 times the frequency in Hertz. Even for these significant sinusoidal accelerations, however, there is no discussion of test requirements. This approach is confirmed by actual DOD spacecraft programs, which rarely perform testing for the low frequency quasi-static environment.

Agency	Reference	Test	Qualification Test Requirements/Typical Levels	Acceptance Test Requirements/Typical Levels
Department of Defense	1	Modal Survey	Required/Low level	Not required
		Shock	Required/Activation of all shock events (Notes 1, 2)	Optional/Single activation of significant shock events
		Acoustic	Required/Max expected + 6 dB, 2 min duration (Note 1)	Required/Max expected spectrum, 1 min dur (Note 4)
		Random Vibe	Required in place of acoustic test for vehicles <180 kg	Required in place of acoustic test for vehicles <180 kg
			Max expected flt + 6 dB, 2 min duration (Notes 1, 3)	Max expected flight, 1 min duration (Notes 3,4)
		Sine Vibe	Not specified/Extreme expected environment (Note 1)	Not specified/Max expected environment (Note 4)
NASA (Goddard Space Flight Center)	14	Modal Survey	Required/Transient, fixed freq, swept sine or random input (note 5)	Not required
		Shock	Required/Actual or simulated shock events (Note 6)	Required/Actual or simulated shock events (Note 6)
		Acoustic	Required/Max expected flight + 3 dB, 2 min duration	Required/Max expected flight, 1 min duration
		Random Vibe	When practicable, required for small payloads (Scout)/	When practicable, required for small payloads (Scout)/
			Max expected flight + 3 dB, 2 min per axis	Max expected flight, 1 min per axis
		Sine Vibe	If necessary/1.25 x max expected flight, 2 oct/min	If necessary/Max expected flight, 4 oct/min
Jet Propulsion Laboratory (JPL)	19, 23, 24, 25, 26	Modal Survey	Required	Not required
		Shock	Required	Requirements unavailable
		Acoustic	Required/95% flight env + 4 dB, 3 min duration	Required/95% flight environment, 1 min duration
		Random Vibe	Required	Requirements unavailable
			2.5 x 95% flight environment, 3 min per axis, 3 axis	
		Sine Vibe	Required/1.5 x 95% flight environment, 4-6 oct/min	Required/1.0 x 95% flight environment, 4-6 oct/min

Notes:

(1) Based on extreme expected environment (P99/90)

(2) One activation of all shock producing events, 2 additional activations of significant events (within 6 dB of envelope of SRS from all sources)

(3) Each of three axis

(4) Based on max expected environment (P95/50)

(5) Required at subsystem level, may be performed at payload or component level, if appropriate

(6) Qual: Two actuations, 1.4 x max expected flight level, 2 x each axis, Acceptance: One actuation, 1 x max expected flight level, 1 x each axis

Table 5.1. Governmental Agency Spacecraft Dynamic Test Requirements.

The MIL-STD-1540C acceptance level dynamic test requirements are similar to the qualification requirements. While the shock test is optional at the acceptance level, the acoustic or random vibration tests are required, again depending on vehicle size. An acceptance modal survey is not required. Levels for shock, acoustic and random vibration acceptance tests are defined as the maximum expected environment, which is the level not exceeded on at least 95 percent of flights, estimated with 50 percent confidence (P95/50). As with the qualification test requirements, acceptance level sinusoidal environments are discussed, but test requirements are not mentioned. In actuality, acceptance level sinusoidal vibration tests are rarely performed on DOD programs.

2. NASA

NASA requirements for the testing of spacecraft and components are detailed in Reference 14, the General Environmental Verification Specification for STS and ELV Payloads (GEVS-SE). The requirements for dynamic testing at the vehicle level are summarized in Table 5.1. The structural verification process begins with the development of a finite element model which is verified by test. The analysis is required to define the payload's modal frequencies and displacements below a frequency dependent on the launch vehicle model. A modal survey may be required if the resonant frequencies of the spacecraft subsystems are not greater than the upper frequency of the model for the launch vehicle of concern.

Qualification of NASA payloads for the vibroacoustics environment requires an acoustics test at the vehicle level. "In addition, a random vibration test shall be performed when practicable to better simulate structure borne inputs." [Ref. 14] For large payloads, the upper frequency of the random vibration test is 200 Hz at a minimum. For small payloads (less than 1000 lb), the random vibration shall be conducted over the full 20-2000 Hz frequency band. Limit levels for vibroacoustic testing are equal to the maximum expected flight environment and qualification levels are specified as the limit levels plus 3 dB. Durations are 2 minutes for qual acoustics testing (or 2 min/axis for random) and 1 minute for protoflight (1 min/axis).

NASA structural verification requirements include the low frequency quasi-static environment if the particular launch vehicle has a significant component in that region. Additional

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vibration tests, such as sinusoidal vibration, shall be performed to qualify the payload for inputs such as sustained oscillations due to MECO and POGO effects. Requirements for sinusoidal or other low frequency testing are to be determined on a case by case basis. If conducted, the sinusoidal vibration test levels are specified as 1.25 times the limit level at 2 oct/min for qualification or 4 oct/min for protoflight. In addition, the payload must be qualified for the shock induced during separation and for any other externally induced shocks not enveloped by the separation shock.

For acceptance, the GEVS-SE specifies that vibroacoustic and other vibration testing shall be conducted at the maximum expected flight levels using the same duration as recommended for protoflight hardware (1 minute for acoustics tests and 1 minute per axis for random vibration testing). If necessary, sinusoidal vibration is conducted at limit levels at a sweep rate of 4 octaves per minute. Shock testing at the acceptance level should be considered on a case by case basis.

3. Jet Propulsion Laboratory

While a detailed environmental test specification from the Jet Propulsion Laboratory (JPL) was not available, information regarding current JPL test practices was found in several papers. As most JPL spacecraft are one-of-a-kind vehicles flying interplanetary missions, the test requirements very stringent. Intensive subsystem testing is performed and "a full formal environmental test program is required." [Ref. 23: p. 157] In general, JPL conducts modal and static testing for each spacecraft program on a Developmental Test Model (DTM). Acoustic testing is conducted on all spacecraft at the qualification and acceptance levels. In addition, a sinusoidal vibration test of one to three axis is performed on qualification or protoflight spacecraft and generally on flight vehicles. [Ref. 24: p. 14] Because the interplanetary spacecraft usually built by JPL are relatively small, vibration testing is much more likely to be performed. A summary of typical JPL requirements is shown in Table 5.1.

An example of the typical JPL environmental testing approach is the Topex/Poseidon program. This program conducted protoflight acoustic testing at maximum expected flight levels plus 4 dB for one minute. Swept sinusoidal vibration testing was also performed on the

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protoflight vehicle in the longitudinal axis with levels of 1.5 g peak at a 4 oct/min sweep rate between 5 and 100 Hz. [Ref. 25] For the Viking Orbiter Developmental Test Model (ODTM), sinusoidal vibration levels were set at 1.5 g peak from 8-200 Hz in the lateral axis and 1.5 g peak from 20-128 Hz in the longitudinal axis (1.0 g peak for flight acceptance). [Ref. 26: p. 12]

Recently JPL has adopted a vertical axis force limited random vibration test at the vehicle level for the Cassini spacecraft. This test is intended to replace the conventional swept sinusoidal vibration test. The random vibration test is specified at a level of .05 g^2/Hz in a band between 5 Hz and 200 Hz. [Ref. 19]

4. Commercial

A variety of approaches to dynamic testing exist in the commercial space industry. Because the space industry is so competitive, commercial manufacturers do not generally provide in-depth information about their test programs. However, some information can be obtained from journal articles and other sources. Most manufacturers perform acoustic and shock testing at the vehicle level. Many also perform sinusoidal vibration testing. Still others perform random vibration and do not perform acoustic vibration, even for larger vehicles. The selection of tests a company performs is subject to a variety of influences including corporate expertise and available facilities.

An illustration of a typical environmental test program for a commercial communications spacecraft is provided by the AUSSAT B spacecraft built by Hughes. The AUSSAT B was the first spacecraft in the HS-601 series of body stabilized communications satellites. The protoflight spacecraft was subjected to both acoustic and sinusoidal vibration testing. Acoustic levels were 3 dB above the expected flight levels while sinusoidal vibration was conducted from 5 to 100 Hz at unknown levels. [Ref. 27]

B. TEST EFFECTIVENESS

As previously discussed, spacecraft level dynamics tests are conducted for two purposes. The first purpose is that of qualification: to ensure that the hardware can survive and operate in the expected environment, as well as to verify the analytical model. The second purpose is that of

acceptance: to provide quality control assurance against workmanship or material deficiencies. Consequently, the effectiveness of dynamics tests must be evaluated with regard to each of these purposes. For qualification, the test must adequately simulate the actual environment without over or under testing the spacecraft. For acceptance, the test must provide an adequate workmanship screen, again without over or under testing.

To the extent practical, this study will evaluate the effectiveness of acoustic, random vibration, sinusoidal and transient tests for both qualification and acceptance. Test effectiveness will be evaluated both qualitatively and quantitatively. As the modal survey is primarily an analytical test, it will not be included in the evaluation. In addition, as shock testing practices are relatively uniform throughout government and industry, an evaluation of shock testing will not be included.

1. Acceptance

a. *Criteria*

To establish the effectiveness of the various tests for acceptance purposes, several criteria can be used. As the goal of acceptance testing is to expose workmanship or material deficiencies in a spacecraft, one suitable criteria for acceptance testing effectiveness is the number of failures detected by a particular test. A test program which exposes more failures per vehicle or a particular test which exposes a greater percentage of failures than other tests can be considered more effective. Several test effectiveness studies have been performed which utilize such measurements of effectiveness. One such study, discussed in Reference 2, quantified test effectiveness as

$$TE = F_1/F_2 * 100$$

where

TE = the test effectiveness for the test of interest

F_1 = total failures found in the test of interest

F_2 = total failures found in all tests and in early flight

For this study, early flight is defined as the first 45 days of flight.

This definition of test effectiveness is based on the "roller coaster" failure rate curve shown in Figure 5.1 and described in Reference 2. Due to infant mortality, the failure rate during test and early flight is higher than during the normal operating period. The failure rate typically decreases after launch, ascent and on-orbit testing, a period which typically lasts about 45 days. It is assumed that the failures found by the acceptance tests would have occurred in early flight had the testing not been performed.

For this study, the measurement of test effectiveness as described above has been applied to data collected from a variety of spacecraft development programs. In addition, a survey was conducted of other test effectiveness studies for comparison to the data collected above. Finally, results of an experimental evaluation of workmanship test effectiveness were analyzed and compared to the various study results.
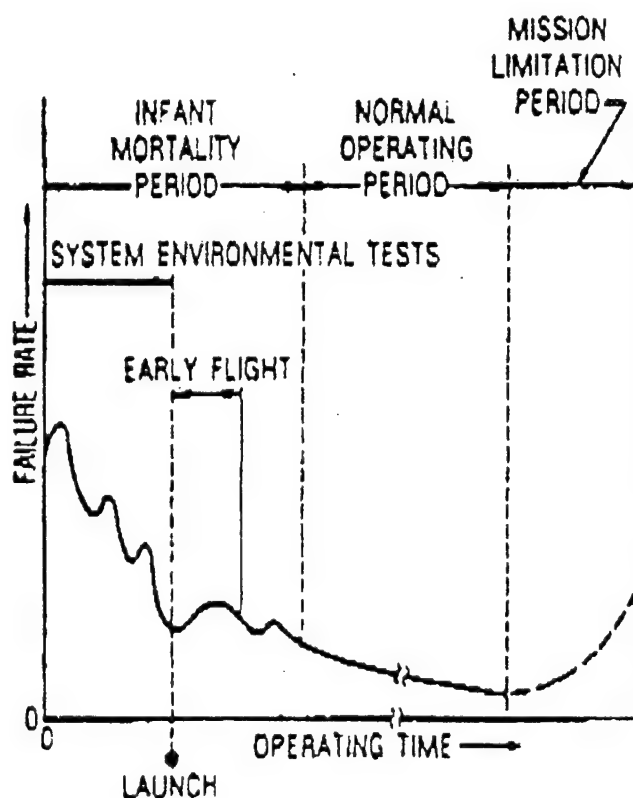
Figure 5.1. "Roller Coaster" Failure Rate Curve. From Ref. [2].

b.   Data Analysis

Data on the dynamic test failures and early on-orbit failures for 17 different spacecraft development programs are presented in Table 5.2. Data were obtained from References 28, 29 and the Space Systems Engineering Database (SSED) at the Aerospace Corp, El Segundo, CA. Tests, test levels and failure rates are presented for a total of 78 satellites. Overall, the effectiveness of dynamics testing at the vehicle level was 15%. Dynamic testing exposed an average of 0.9 failures per satellite and the early flight failure rate was 0.7 failures per satellite.

Most of the data shown in Table 5.2 is from DOD programs which perform only acoustic acceptance testing at the vehicle level. However, 4 programs conducted additional dynamics testing including sinusoidal vibration and in one case, sinusoidal and random vibration. The data for these programs, which includes a total of 23 satellites, does not indicate any advantage to the performance of the additional dynamics tests. While the number of test failures detected per vehicle was higher for the programs which performed the additional tests (1.5 dynamic test failures per satellite versus 0.9 acoustic only test failures per satellite) the inflight failure rates were not reduced, indicating that the additional testing did not appreciably improve the on-orbit performance of the spacecraft concerned. This conclusion is supported by the fact that of the 23 spacecraft which were exposed to the sinusoidal and/or random vibration dynamics tests, only one failure was attributable to those additional tests. By far, the vast majority of failures were found during the acoustic test, indicating that acoustic testing was the more perceptive test.

There are several factors which could impact the number of failures found during each of the dynamic tests. The first of these factors is the order in which the tests were performed. In three out of four of the programs which performed acoustic and sinusoidal or random vibration testing, the acoustic tests were performed first. The fact that the acoustic testing exposed most of the dynamic-related test problems could therefore be attributable only to the fact that the acoustic tests had the first opportunity to expose the defects. Unfortunately, failure data for the one program that did perform sinusoidal vibration first did not separate failures by the specific dynamic test in which they occurred, so no conclusions could be drawn from the data for that program.

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Program	No of Satellites Tested	Test Failures/Satellite			Test Levels /Duration		No of Satellites Flown	Flight Failures per Satellite
		Acoustic	Sine Vibe	Random Vibe	Acoustic (dB)	Vibration		
E2	4	0.5			143/1 min		4	0.5
D1	3	0.3			142/1 min		3	2
D2	1	0			142/1 min		1	1
D3	9	0.9			142/2 min		7	0.6
D4/D5	2	0.5			145/148/139 (1)		1	0
B	16	0.6	0		142/1 min		11	0.6
G	4	1			139/1 min		3	2
F1	5	0.4			139/1 min	Note (2)	4	0.3
F2	3	0.7			139/1 min		1	0
H1	2	0.5			140/1 min		2	1
H2a	1	2			140/1 min		1	1
H2b	2	0.5			140/1 min		2	0.5
C	8	1.1			145/1 min		7	0.4
A	12	1.1	NA	NA	NA (3)/1 min	Note (4)	9	NA
I	3	1	0.3		NA/1 min	Note (5)	3	NA
J	3	3.7	NA	NA	138/1 min	Note (6)	3	NA
Total	78	0.9					62	0.7

Notes:

(1) D4: 145 dB for 2 min, D5: 148 dB for 1.5 min/139 dB for 12 min

(2) Sine: 0.45 g peak, 5-100 Hz, longitudinal axis

(3) NA = Not Available

(4) Sine: 0.3 g peak, 10-500 Hz, 2 oct/min, longitudinal axis

   Random: 2.04 grms, 1 min, longitudinal axis

(5) Sine: Design limit, 2-50 Hz, 2 oct/min

(6) Sine: 1.0 g peak, 10-200 Hz, 6 oct/min

Table 5.2. Dynamic Test and Flight Failure Data.

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Additional factors which could impact the number of failures found by each particular dynamic test are the relative magnitudes and durations of the tests. For two of the programs which performed sinusoidal vibration tests, these tests were conducted at the relatively low levels of 0.3 g and 0.45 g peak. Using the equivalency formula from Chapter IV, Section C.1, these levels correspond to 0.007 g^2/Hz and 0.015 g^2/Hz equivalent random inputs at 50 Hz and Q = 20, respectively, which is well below the typical random vibration input levels specified in the MIL-STD-1540C. While program J used a more conservative sinusoidal vibration input of 1.0 g peak, corresponding to 0.074 g^2/Hz random vibration input at 50 Hz, the sweep rate of 6 oct/min was slightly more rapid. Applying the test duration relationship discussed in Chapter IV, Section C.2, this corresponds to a factor of 1/3 fewer cycles at a particular resonance frequency, which results in a significantly lower impact.

While some preliminary conclusions can be drawn from the data presented above, it is clear that there is not enough information on programs which have conducted sinusoidal and random vibration to provide ant statistical significance. As discussed previously, while test failure data is easily available for most DOD programs, these programs tend to rely solely on acoustic testing as a dynamic acceptance test at the vehicle level. Many of the commercial programs conduct sinusoidal vibration tests for acceptance, but because the commercial spacecraft manufacturing business is so competitive, it is difficult to obtain test results for commercial programs. Finally, data is difficult to obtain even for the NASA and JPL programs which regularly conduct sinusoidal and random vibration testing at the vehicle level. While this data probably exists, it is not regularly published in papers or reports and the particular agencies seem hesitant to release specific data. While an effort is underway to add NASA and JPL test and flight failure data to the Space Systems Engineering Database (SSED) managed by the Aerospace Corporation for the Air Force, the addition of this data is at least a year away.

c. ***Survey Results***

A variety of other test effectiveness studies were surveyed to provide a comparison to the data discussed above. These studies are summarized in Table 5.3. Once again, there is a lack of data from programs which made consistent use of sinusoidal and random vibration in

Agency/Manufacturer	Data Source	Total Spacecraft	Dynamics Tests Performed/Scope of Data	Data Presented	Analysis/Comments from Reference
Ford (Loral)	Ref. 30	28	Acoustic and vibration/Environmental test failure data	Acoustics related = 0.6% of test failures Vibration related = 2.5% of test failures	"Vibe effective workmanship screen" "Acoustics not of benefit"
LMSC	Ref. 31	71	Acoustic only/Acceptance test and flight failure data	653 total test/flight failures, 118 flight, 104 acoustic related	Approx 1.5 acoustic failures/veh Acoustic=15.9% of all test failures "Acoustic best workmanship screen"
LMSC	Ref. 10	23	Acoustic only/Test failure data only	346 total test failures 26 failures in acoustic/post acoustic	Approx 1.0 acoustic failures/veh Acoustic=7.5% of all test failures
LMSC	Ref. 32	81	Acoustic only/Acceptance test and flight failure data	64 failures due to acoustics 7 first day flight failures (Dynamic?)	Approx 0.79 acoustic failures/veh
USAF Space Division	Ref. 2	55	Acoustic only/Test and flight failure data	4.2 total failures/veh, 0.73 acoustic/veh 0.74 early flight failures/veh (1)	Approx 0.73 acoustic failures/veh Acoustic=15% of all test and flight failures
USAF Space-Division	SSED	15	Acoustic, sine and random vibration/Test failure data only	113 total failures, 17 dynamic	Approx 1.1 dynamic failures/veh Dynamics=15% of all test failures
FleetSatCom	Ref 41/SSED	5	Acoustic and sine vibration/Test failure data only	44 total failures, 6 acoustic, 0 sine 20 functional failures	Approx 1.2 dynamic failures/veh Dynamics=13.6% of all test failures
NASA	Ref. 28/29	Galileo/Voyager	Acoustic and sine vibration Test failure data only	102 total failures, 11 dynamic Galileo: rand=27.5%, sine=67%, acoustic=3% of dynamic failures	Approx 3.7 dynamic failures/veh Dynamics=11% of all test failures Vibe: 1 axis=33%, 2 axis=100% of failures detected in vibration

Notes: (1) Early flight failure defined as failure in first 45 days of mission

Table 5.3. Workmanship Test Effectiveness Survey.

vehicle level acoustic testing. One study which does include data regarding vibration testing was done by former Ford Aerospace and Communications Corporation [Ref. 30], now Loral Space Systems Division. According to the Ford Aerospace report, "vibration testing has consistently provided an effective screen for workmanship defects. On the other hand, acoustics testing has not been of benefit as an acceptance test." [Ref. 30: p. 65] The report attributes 2.5% of the total system level acceptance test failures to vibration testing and only 0.6% to acoustics. The total of 3.1% dynamic test related failures out of all system level tests is lower than other studies, which typically associate between 10% and 15% of test failures with dynamics testing.

There are a variety of factors which could impact the results of the Ford study. These include program maturity, the order in which the dynamics tests were performed and the test levels and durations. For more mature programs, such as the communications satellites often manufactured in the commercial arena, failure rates can go down as multiple vehicles of the same design are built. In addition, as with the discussion of the previous data, more failures could be detected by a particular dynamic test just by the fact that it was performed first. Finally, as discussed before, relative test levels and durations can be a factor. Because no further details are available regarding the specific tests conducted by Ford, further conclusions cannot be made.

Similar studies performed by Lockheed Missiles and Space Company (LMSC) are summarized in Table 5.3. As discussed in References 10, 31 and 32, the LMSC results are comparable to those resulting form the DOD data. Like the DOD programs, the LMSC study discussed programs which employed acoustic testing as the only spacecraft level dynamics test. The LMSC data included test results for up to 81 spacecraft and indicated that acoustic testing detected approximately 15% of all test and flight failures with an average of between 0.7 and 1.0 acoustic related failures per vehicle. These results are very consistent with data from the SSED.

d.   Experimental Results

The final source of information on the effectiveness of spacecraft level acceptance tests consists of experimental studies. While such experiments are rare, a recent effort by JPL provides some useful results. As described in Reference 33, this particular experiment used a

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flight spare Voyager spacecraft structural section mounted on an electrodynamic shaker table. Attached to the structure were a number of cantilever beams with natural frequencies in the vertical plane between 10 and 103 Hz. The beams were attached to a backing plate by a cap screw which was torqued to a specified level. A mechanical defect was represented by the improper torquing of the cap screw. If the screw loosened during the vibration test, the associated beam would rotate, indicating a mechanical fault. As the beam rotation was highly visible, the exact number of faults occurring as a function of elapsed time could be determined. In addition, electrical defects were simulated by a 140 bulb light string. An electrical fault occurred when the connection to the bulb was loosened, resulting in a loss of current to the element.

The JPL experiment was performed for various swept sinusoidal and random vibration levels and durations and also with a transient pulse input. Test results are summarized in Table 5.4. The experiment indicated that random vibration, when conducted at levels of at least 0.05 g^2/Hz detected the greatest number of possible defects. The swept sinusoidal vibration test, conducted at a level of 1.5 g peak at a sweep rate of 2 oct/min detected less than half the faults detected by the 0.05 g^2/Hz random vibration test. The lower level random vibration tests (0.01 and 0.04 g^2/Hz) also produced less than half the defects produced by the 0.05 g^2/Hz random vibration test. No electrical or mechanical faults were detected by the transient test. This result was not surprising as the transient pulse duration was only 1.0 second with a peak level of under 2 g's.

Test Type	Test Levels	Results
Swept Sine Vibration	Level: 1.5 g peak (1.5 x Max environment of 95% of flights) 5-200 Hz, 2 oct/min (2.6 min duration), vertical axis Test run with and without force limiting	Force limited: 2 runs, detected 20-30% of all possible mech failures detected < 1% of possible electrical failures Non-force limited: 2 runs, detected 25-50% of all possible mech failures detected 1-5% of possible electrical failures
Random Vibration	Levels: 0.001 g^2/Hz @ 10Hz, +3dB/oct to 100, 0.01 g^2/Hz to 150Hz 0.005 g^2/Hz @ 20Hz, +3dB/oct to 150, 0.04 g^2/Hz @ 150Hz 0.05 g^2/Hz, 5-200 Hz, with and without force limiting Up to 8 min duration, vertical axis	0.01 g^2/Hz: 25% of possible mech failures detected after 6 minutes 0.04 g^2/Hz: 35% of possible mech failures detected after 9 minutes 0.05 g^2/Hz, force limited: 75% of poss mech failures det after 8 min detected < 2-8% of possible electrical failures 0.05 g^2/Hz, non-force lim: 80% of poss mech failures det after 8 min detected 10-25% of possible electrical failures
Transient	Level: Approx 2 g peak 1.0 second duration, vertical axis.	No mechanical or electrical failures detected in 5 runs

Table 5.4. JPL Workmanship Experiment Results.

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A plot of the number of failures detected versus time for the swept sinusoidal and random tests is shown in Figure 5.2. The plot indicates that the effectiveness of the random vibration test peaks after approximately 2.0 minutes. Additional random vibration exposure does not reveal a significant number of additional faults. This result is clearly useful for determining the maximum duration of acceptance level vibration tests.
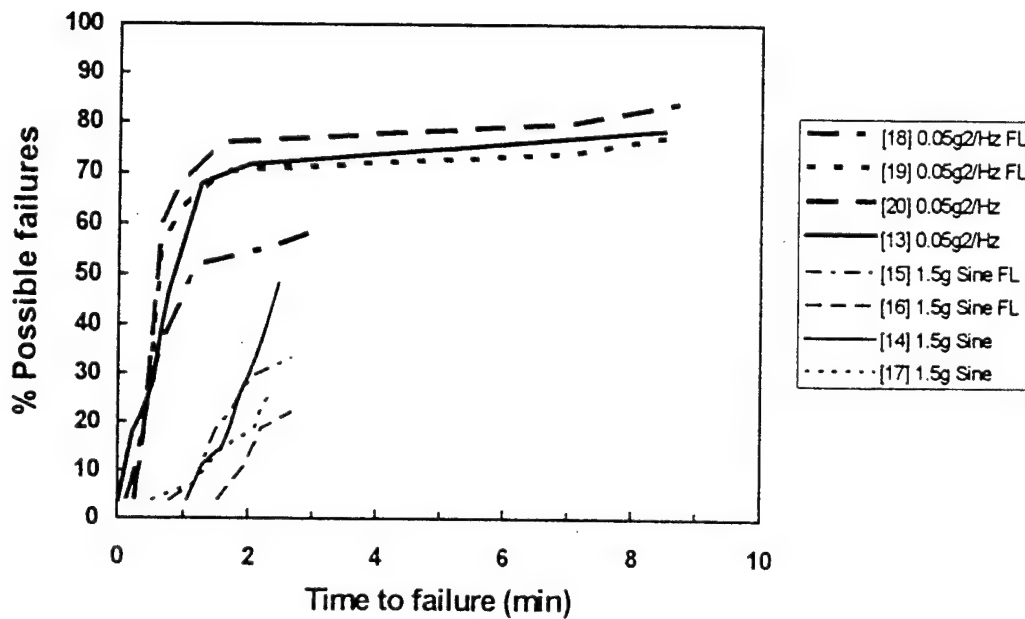
Figure 5.2. Workmanship Experiment Failure Data. From Ref. [33].

The JPL results show that the random vibration test, when performed at the proper level, is more effective than either the swept sinusoidal or transient tests for the exposure of mechanical and electrical defects at the vehicle level. As the overall size and weight of the test vehicle in the JPL study was not indicated, the applicability of this experiment to different size vehicles is unknown. While this experiment shows that the random vibration test was effective for the given vehicle, the acoustic test may still provide benefits for larger vehicles.

An interesting contrast to the JPL experiment is offered by data collected for two recent thesis produced by students at the Naval Postgraduate School. [Refs. 34 and 35] This data

provides the opportunity to compare low frequency swept sinusoidal vibration tests of a large space structure with low frequency random vibration tests of the same structure. While the collection of workmanship failure data was not a part of this test, the response data collected does provide worthwhile information regarding the relative severity of the two test inputs. Such information is important with regard to the potential of over testing.

The sinusoidal and random vibration tests of concern were performed on the Topaz II space nuclear power system at the Sandia National Laboratory Vibration Test Facility in Albuquerque, New Mexico. The Topaz II structure is 3.9 meters tall with a total mass of 1061 kg, which is comparable to the size of a small spacecraft. The structure was instrumented with 22 accelerometers placed in a variety of locations to monitor response of the vehicle in all three axis.

The input sinusoidal and random vibration spectra are listed in Tables 5.5 and 5.6. The maximum random vibration level of 0.06 g^2/Hz is slightly higher than the maximum of 0.05 g^2/Hz used in the JPL workmanship test while the maximum sinusoidal vibration level of 1.0 g is lower than the 1.5 g's applied in the JPL experiment. Duration of the random test was 1 minute with a frequency band of 20 to 2000 Hz. The sweep rate for the sinusoidal test was 0.25 oct/min across a frequency range of 5 to 200 Hz which is slower than the 2 oct/min sweep rate employed in the JPL test. While the slower sweep rate would result in higher responses at the resonance frequencies of the structure, the lower input would offset this to an extent.

FREQUENCY (Hz)	LEVEL (g)
5	0.25
5-8	linear increase
8-40	1.0
40-100	0.9
100-200	0.8

Table 5.5. Topaz II Sinusoidal Vibration Test Inputs.

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FREQUENCY (Hz)	PSD (g^2/Hz)
20-70	0.02
70-100	linear increase
100-800	0.06
800-2000	linear decrease
2000	0.013

Table 5.6. Topaz II Random Vibration Test Inputs.

Responses for the random and sinusoidal inputs can be compared in the axial direction for those natural frequencies which fall within the range of both tests (i.e.-20 to 200 Hz). This omits the first axial mode which in this case provided a lower response than both the second and third modes in most accelerometer locations. The data, which is presented in Table 5.7, indicates that the response to the sinusoidal vibration input is up to 100 times greater than the response to the random input. While no predicted or maximum flight response amplitude values were available for comparison, the magnitude of the sinusoidal response, which reaches 11 dB at one location, is clearly high enough to point out the potential for over test. The random test, however, results in a relatively mild response. This is an interesting contrast with the results of the JPL workmanship test. While the sinusoidal testing clearly subjects the structure to much larger responses, the random test was shown to be more effective in uncovering workmanship and material defects.

2. Qualification

a. Criteria

The goal of qualification testing is to ensure that the hardware under test can survive and operate in the expected environment, as well as to verify analytical models. Consequently, an analysis of the effectiveness of qualification tests must focus on the ability of the test to simulate the environment in which the spacecraft must operate. The test should therefore accurately

Location	Frequency (Hz)	Sine Response (g)	Random Response (g)	Response Ratio
Reactor Leg Bracket (I08/C1)	39.18	3.21	0.146	22.0
	119.17	0.84	0.054	15.6
Reactor Plenum (I10/C02)	43.75	1.47	0.049	30.0
	75.44	0.22	0.017	12.9
Reactor Top Plenum (I10/C03)	39.18	4.51	0.243	18.6
Reactor Leg Bracket (I07/C05)	40.00	6.00	0.201	29.9
	111.37	0.55	0.014	39.3
Reactor Top Plenum (I09/C09)	35.37	6.14	0.193	31.8
Reactor Top Plenum (I09/C12)	39.48	0.99	0.025	39.6
Leg Base (I01/C13)	39.50	2.36	0.020	118.0
Leg Base (I02/C16)	38.06	1.69	0.022	76.8
Leg Top (I06/C19)	39.50	5.00	0.079	63.3
	107.44	0.67	0.024	27.9
Bottom Collector (I04/C24)	38.45	6.20	0.076	81.6
	112.57	1.62	0.116	14.0
Bottom Collector (I03/C27)	38.45	4.70	0.085	55.3
	109.03	1.39	0.075	18.5
Cesium Unit (I11/C28)	39.80	11.09	0.193	57.5
	123.50	2.03	0.018	112.8
Leg Joint (I05/C31)	38.56	3.58	0.044	81.4
	101.86	0.69	0.024	28.8
Startup Unit Frame (I12/C34)	39.60	8.75	0.181	48.3
	123.98	0.87	0.013	66.9
Startup Unit Frame (I12/C36)	34.06	3.08	0.078	39.5

Table 5.7. Topaz II Sinusoidal and Random Vibration Test Comparison.

simulate the actual or predicted flight load environment with regard to input magnitudes, frequencies and durations or number of cycles imposed on the vehicle. More importantly, the magnitude of the vehicle responses must be comparable to predicted or actual flight responses. As with acceptance, this must be accomplished without over or under testing and must be as cost effective and easy to perform as possible.

Evaluations of the various dynamic qualification testing methods currently used have been performed using a variety of criteria. These criteria include the peak response levels for the given test type and input and the number of peak cycles to which the test article is exposed. In addition, such characterizations as the shock response spectrum have been used to compare the responses of the various test inputs. For this analysis, a survey of dynamics qualification test experiments and studies using these types of criteria was conducted to determine if a definitive conclusion could be supported regarding the effectiveness of the various qualification test methods. Results of each of the applicable studies will be presented along with conclusions and recommendations for further work.

b. Study Results

(1) Swept Sinusoidal, Transient and SRS Comparison. The data discussed in this section was generated as part of a larger study conducted by MBB/Erno for Intelsat. For this experiment, a single axis test was performed in the axial direction to compare transient, sinusoidal and shock response spectrum excitations to determine the differences between tests using these input types. As presented in Reference 36, the specific inputs were:

1. a sinusoidal input of 1.25 g from 6 to 30 Hz and 0.5 g from 30 to 100 Hz. This input provided the typical uni-axial sinusoidal sweep simulation of environmental loading as specified by launch vehicle user manuals. The reduction of the input above 30 Hz was done to reduce the high loading on the antenna interface of the test model without requiring notching. The results could then be analytically scaled up. applying different notch criteria to produce a common basis for comparison.

2. the transient input shown in Figure 5.3.

3. a shock input as shown in Figure 5.4. This shock response spectrum was built up from the acceleration time history of the transient input using a Q of 50.

To compare the responses generated by the different input excitations, the minimum and maximum values for each degree of freedom were presented together in bar chart from as shown in Figures 5.5 through 5.7. The measured bending moments and interface forces at the spacecraft separation plane were also plotted as shown in Figures 5.8 through 5.10. Three different plot presentations are provided. The plots are the same except for the sinusoidal responses which have been altered by the theoretical application of the different notching criteria. These criteria, which result in the notched inputs corresponding to those indicated in Figure 5.11, are:

A. unnotched input

B. notching against longitudinal forces (bending moments) at the spacecraft/launch vehicle interface.

C. notching against local design loads of particular components such as the antenna, tank and side panel in the axial direction.

D. notching with regard to local responses in all three axis.

The data plots show that the response accelerations for the sinusoidal tests vary depending on the notching criteria, as would be expected. The sinusoidal input clearly results in over testing for all three notching criteria. Sinusoidal responses were obtained ranging from the same order of magnitude of the transient response up to over test levels of 3 times the desired magnitude. Because the notching is performed relative to only a few structural locations, other locations could be under tested. While no significant under testing was observed in this case, this was probably due only to the decoupled structural nature of the test hardware. This condition is an exception and cannot be assumed for usual cases.

The data plots also indicate that the structural responses from the shock test are lower in general than the reference values from the transient test. However, correct loading depends on the reference point for the shock spectrum and can be achieved only for a specific point

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on the structure. Consequently, the transient test resulted in the most realistic simulation of the flight response.
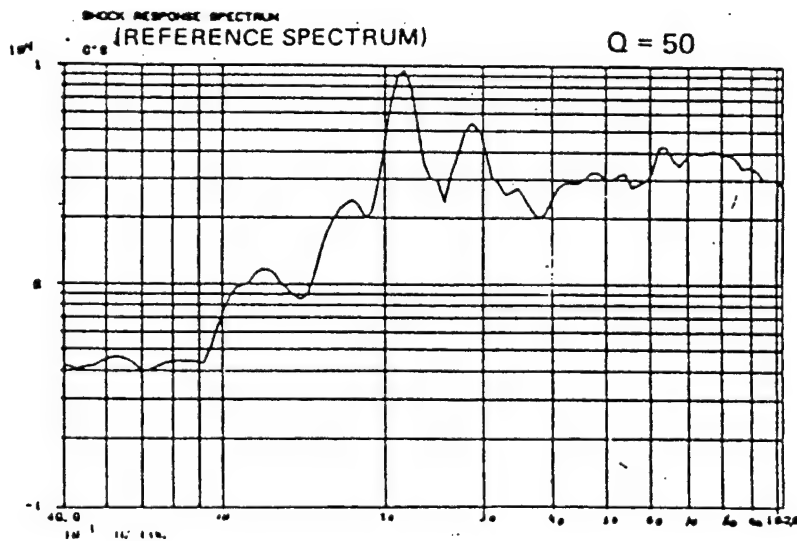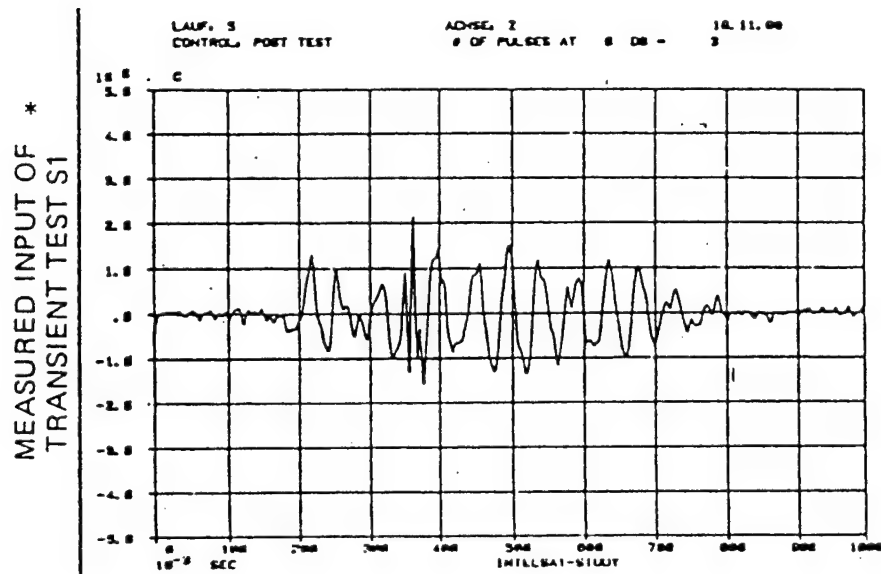
Figure 5.3. Transient Test Input. From Ref. [36].
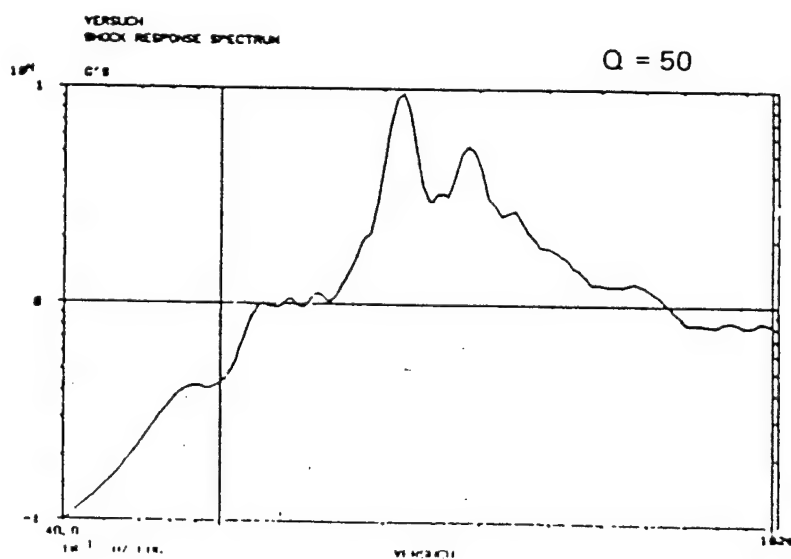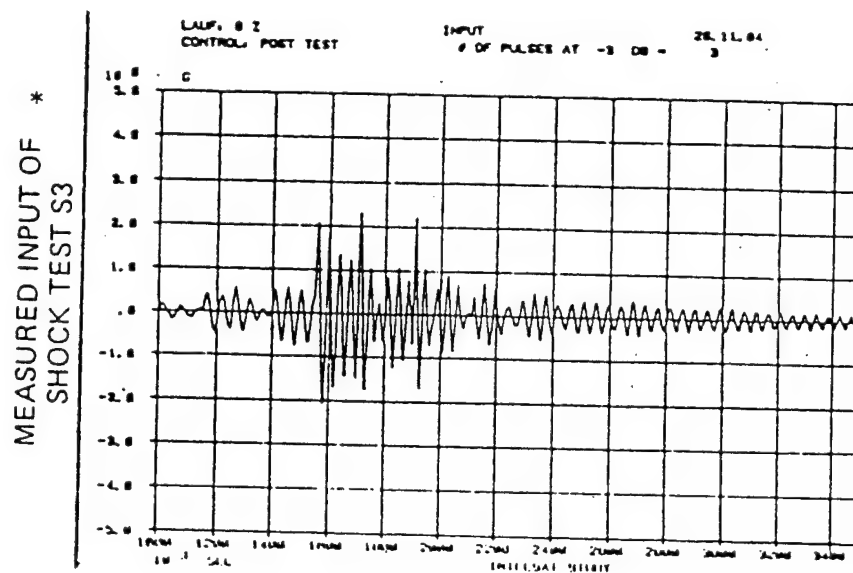
Figure 5.4. Shock Test Input. From Ref. [36].
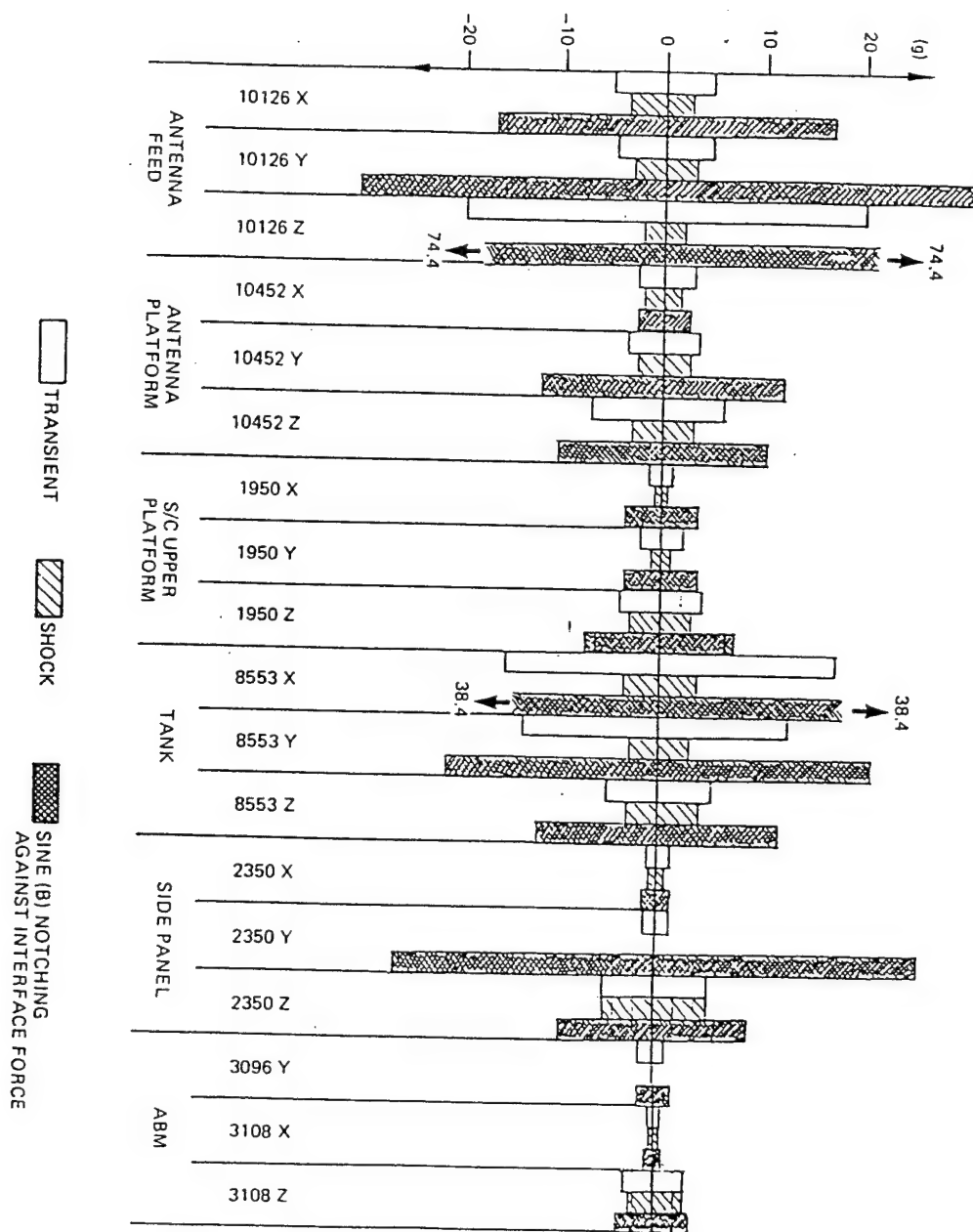
Figure 5.5. Comparison of Response Magnitudes for Notching Criteria B. From Ref [36].

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MIN/MAX RESPONSES DUE TO SINE/TRANSIENT AND SHOCK EXCITATION
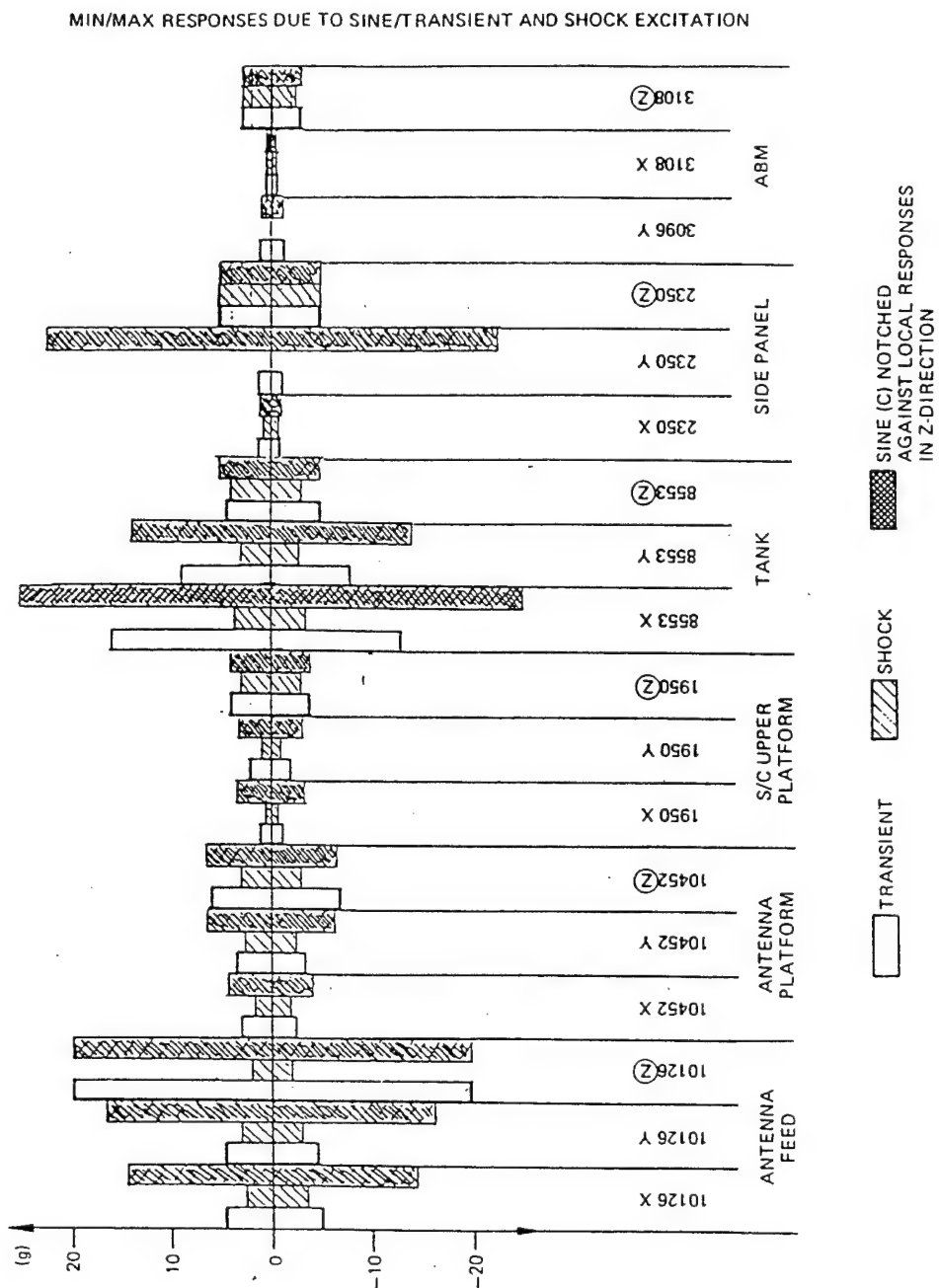
Figure 5.6. Comparison of Response Magnitudes for Notching Criteria C. From Ref [36].
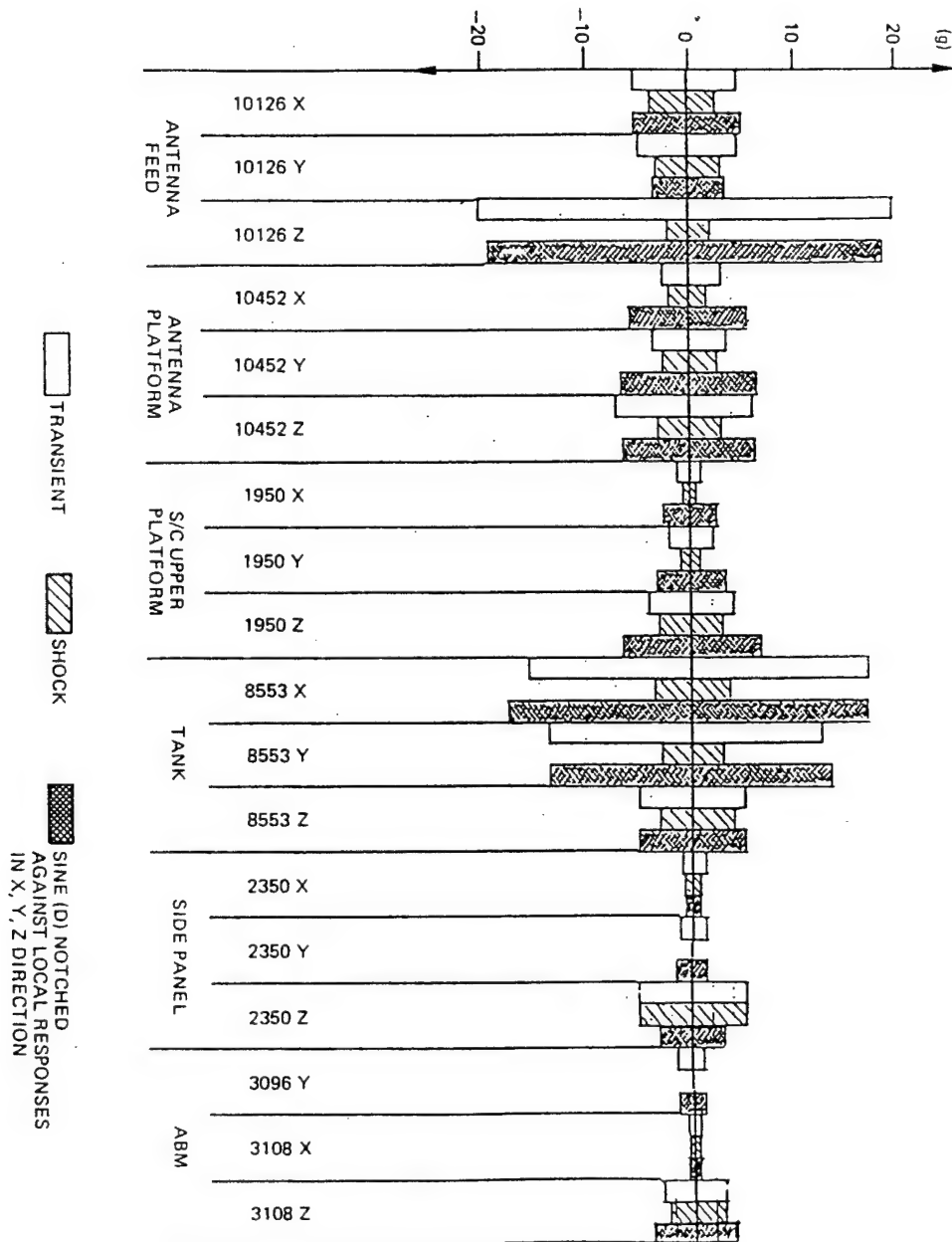
Figure 5.7. Comparison of Response Magnitudes for Notching Criteria D. From Ref [36].
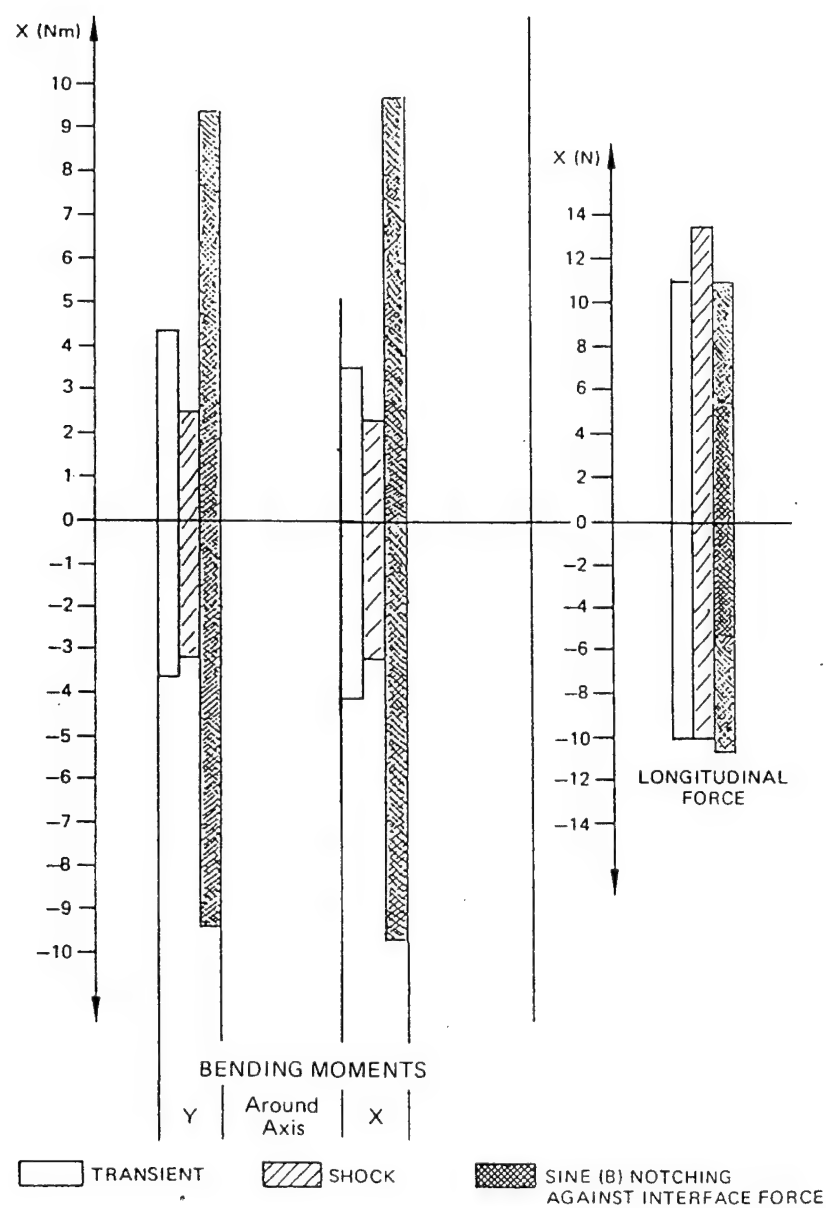
Figure 5.8. Comparison of Bending Moments for Notching Criteria B. From Ref [36].

MIN/MAX RESPONSES DUE TO SINE/TRANSIENT AND SHOCK EXCITATION
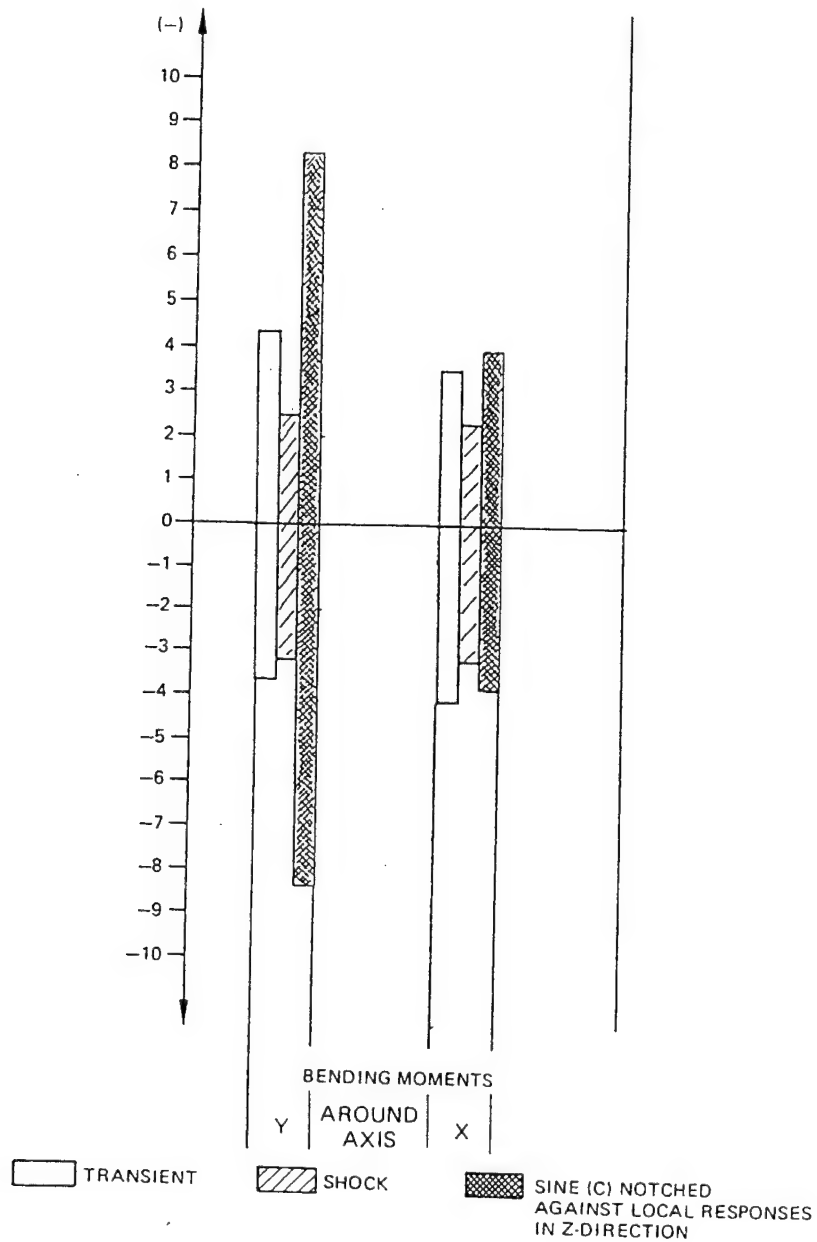
Figure 5.9. Comparison of Bending Moments for Notching Criteria C. From Ref [36].

MIN/MAX RESPONSES DUE TO SINE/TRANSIENT AND SHOCK EXCITATION

BENDING MOMENTS

Y | AROUND AXIS | X |

TRANSIENT   SHOCK   SINE (D) NOTCHED
                        AGAINST LOCAL RESPONSES
                        IN X, Y, Z DIRECTION
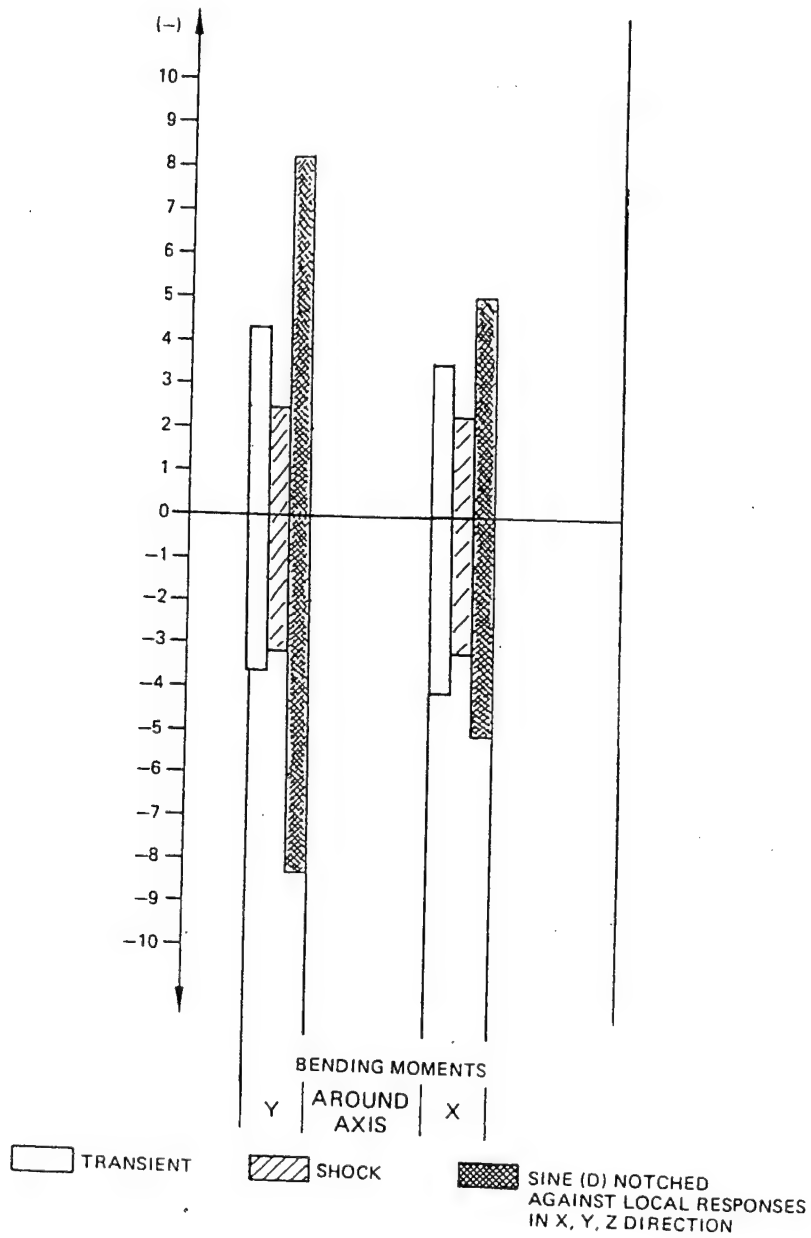
Figure 5.10. Comparison of Bending Moments for Notching Criteria D. From Ref [36].

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A — SPECIFIED INPUT IN Z AT POINT 99
— — — REALIZED INPUT IN Z AT POINT 99

B — NOTCH CRITERIA :
LONGITUDINAL INTERFACE FORCE → 2850 N
AT 40 to 46 Hz MEASURED DURING TRANSIENT
TESTING

MEASURED
LONGITUDINAL
FORCE AT S/C
INTERFACE

C — NOTCH CRITERIA :
w.r.t LOCAL RESPONSES IN Z-DIRECTION
AT POINTS 2350/8553/10126

ONLY Z
RESPONSES

10126

NOTCH CRITERIA :
w.r.t. LOCAL RESPONSES IN
X, Y, Z-DIRECTION
AT POINTS 2350/8553/10126

RESPONSES IN
X, Y AND Z-
DIRECTION

8553

2350

99

D

Figure 5.11. Notching Criteria. From Ref. [36].

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(2) Swept Sinusoidal, Transient and Synthesized Waveform. As discussed in References 20 and 37, laboratory testing was conducted at JPL to measure the relative conservatism between vibration tests using several different inputs. These inputs included a flight-transient excitation, swept sinusoidal and a synthesized waveform with the same SRS as the flight waveform. Single axis tests were conducted on a dynamic mass model of a radioactive thermoelectric generator (RTG) used in recent spacecraft. Several characterizations of the vibration response of the structure were measured and compared. These characterizations included peak acceleration levels and cycles, the rms acceleration as a function of time (TRMS), the rms acceleration as a function of frequency (FRMS) and the shock response spectrum (SRS).

The input signals for these tests are shown in Figures 5.12 through 5.14. A plot of the number of acceleration peaks exceeding a specific amplitude for each test input along with flight data is shown in Figure 5.15. As expected, the swept sinusoidal input provides a much greater number of peaks at a given amplitude than the flight data. In addition, the sinusoidal input registers much higher maximum peak responses with significant numbers of peaks over 8 g's. The transient test most closely follows the flight characterization, though the synthesized waveform is close, resulting in only a small number of peak responses greater than the flight characterization.

Plots of the TRMS and FRMS characterizations for the transient and synthesized waveforms are shown in Figures 5.16 and 5.17. As the duration of a test must be the same to provide comparisons using these characterizations, data for the sinusoidal test was not included. The TRMS plot indicates that the synthesized waveform results in moderate over testing between 0.2 and 0.4 seconds while the transient input results in a slight under test throughout the one second duration. The FRMS characterization, shown in Figure 5.17, shows that both the transient and synthesized waveforms under test across the frequency range from 10 to 100 Hz. Because the transient and synthesized waveforms can both be shaped to provide the response desired, these variations from the desired response can be reduced.

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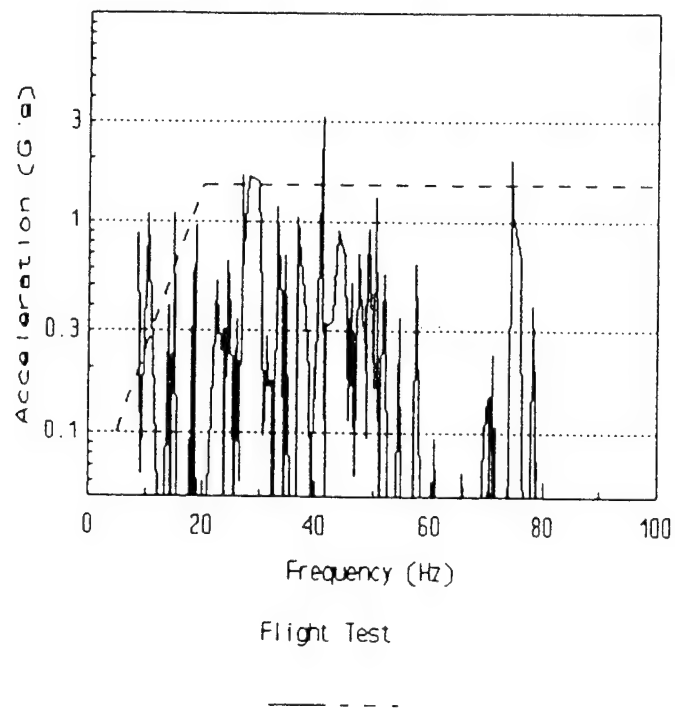
Figure 5.12. Swept Sine Test and Flight Levels. From Ref. [37].
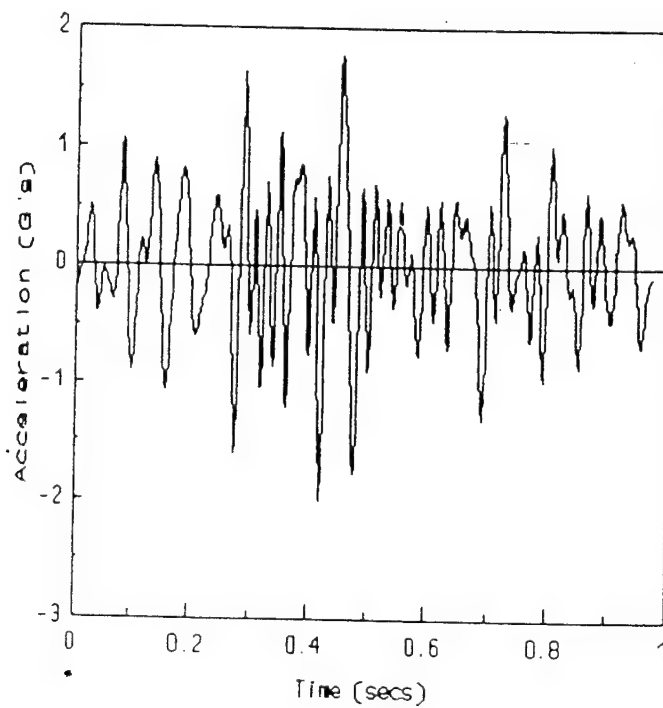
Figure 5.13. Transient Test and Flight Levels. From Ref. [37].

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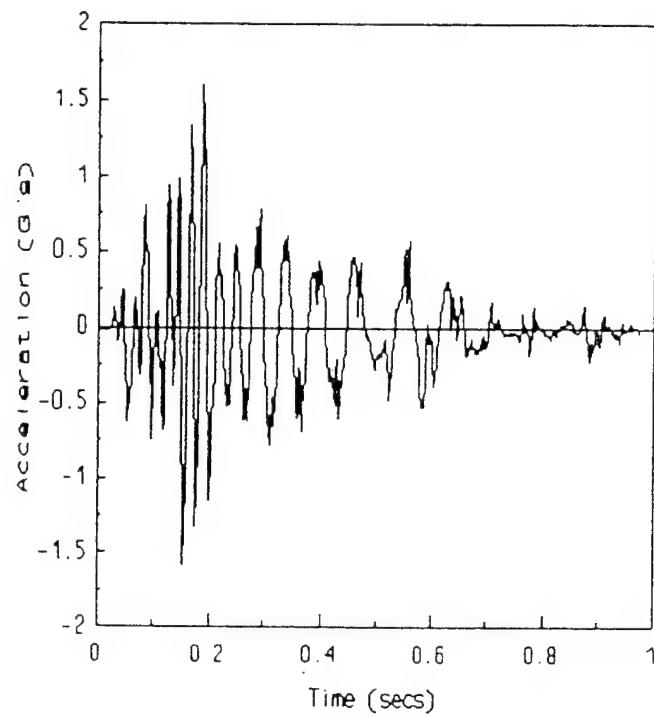
Figure 5.14. Synthesized Waveform Test and Flight Levels. From Ref. [37].
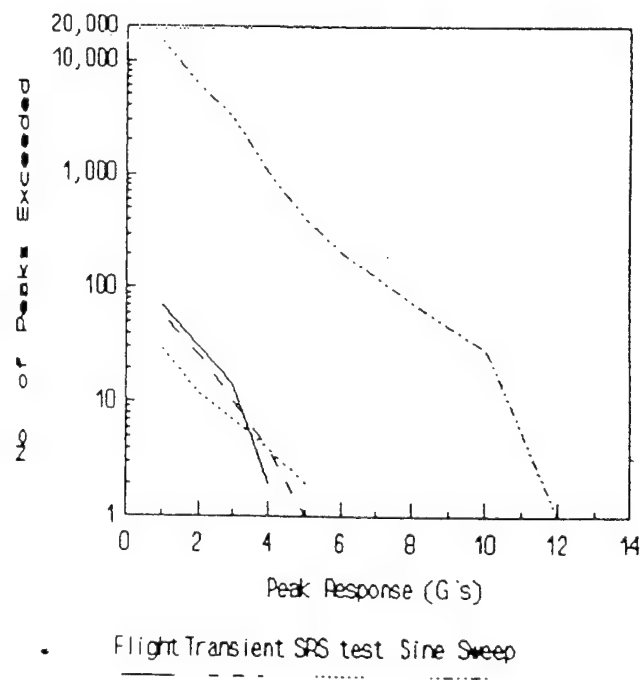
Figure 5.15. Number of Acceleration Peaks Versus Amplitude. From Ref. [37].
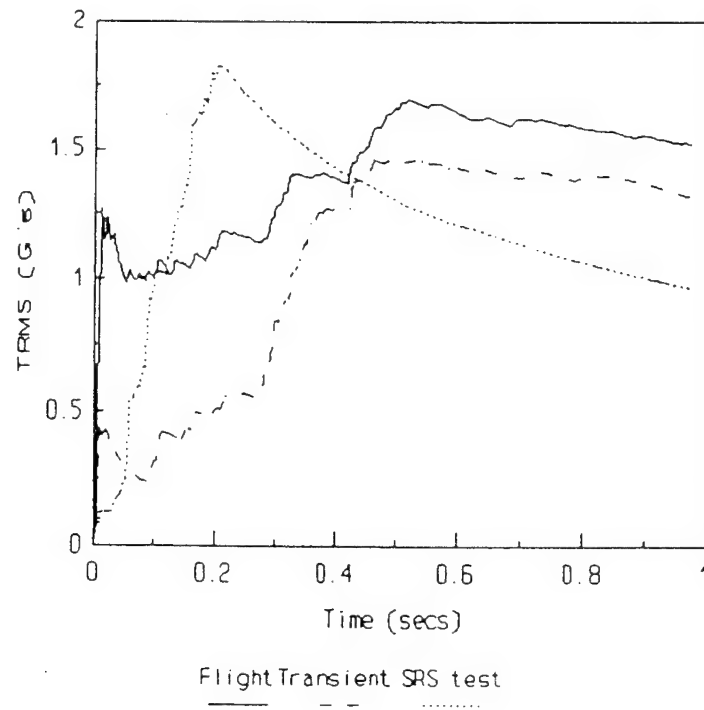
Figure 5.16. RMS Acceleration as a Function of Time. From Ref. [37].
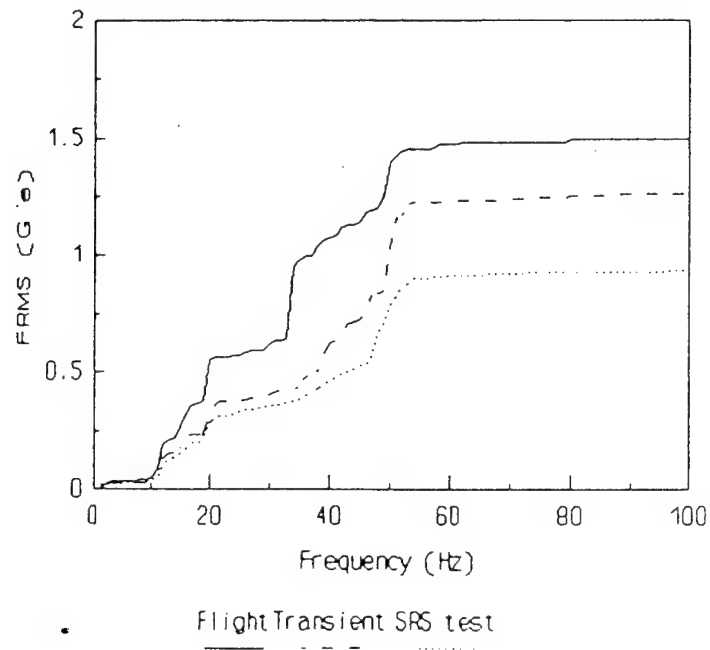
Figure 5.17. RMS Acceleration as a Function of Frequency. From Ref. [37].

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(3) Swept Sinusoidal, Random Dwell and Transient Comparison. This test program, described in Reference 18, was initiated by Intelsat and conducted by Hughes to examine alternatives to avoid the potential for over test inherent with notched swept sinusoidal testing. The test objective was to subject the spacecraft to loads and responses representative of the flight environment with respect to primary and secondary structure. At the time of the test, Intelsat typically specified swept notched sinusoidal testing for the lower frequency range below 75 Hz, which typically envelopes all primary and most secondary structural responses.

For this particular study, seven distinct tests were performed using a prototype Intelsat IV-A vehicle:

1. baseline 4 oct/min notched swept sinusoidal input at 3/4 acceptance level.

2. 8 oct/min notched swept sinusoidal input intended to reduce the number of cycles in test.

3. 4 oct/min notched swept sinusoidal with reduced control input and notch levels identical to those used in the previous 4 oct/min test. The intent of this input was to provide a narrower frequency region over which a resonance would notch, reducing the number of load cycles and thus reducing the secondary response

4. narrow band random dwell consisting of a 3 Hz bandwidth signal at three structural resonances (7, 19, 23 Hz). For this test the input level was raised until the most critical location reached the maximum expected flight response level.

5. narrow band random dwell with broad band background floor. This test was intended to excite secondary resonances at different frequencies from the primary resonances.

6. direct transient reproduction at the spacecraft shelf. This test used a synthesized base excitation which reproduced the transient response at the spacecraft shelf as described in Chapter IV Section B.6.

7. least favorable response at spacecraft shelf. This method attempts to establish a transient base input which guarantees the maximum response at the spacecraft shelf.

As a basis for comparison, flight data was obtained for the Atlas Centaur as discussed in Chapter IV, Section A.3. Cumulative amplitude histograms from a variety of accelerometers provided information on the number of times the response exceeded a given level. The histograms therefore provided information on the number of loading cycles as well as magnitude. Comparative histograms for the flight and test data are shown in Figures 5.18 through 5.20.

Results of the 4 oct/min baseline sinusoidal sweep test, shown in Figure 5.18, show an excessive number of counts at high values of peak moment, which is clearly not desirable. The reduced input 4 oct/min test results in a slight decrease in the number of exceedances at a given level of peak moment but the 8 oct/min test shows the best correlation to flight data in both number of cycles at high values of peak moment and general histogram distribution.
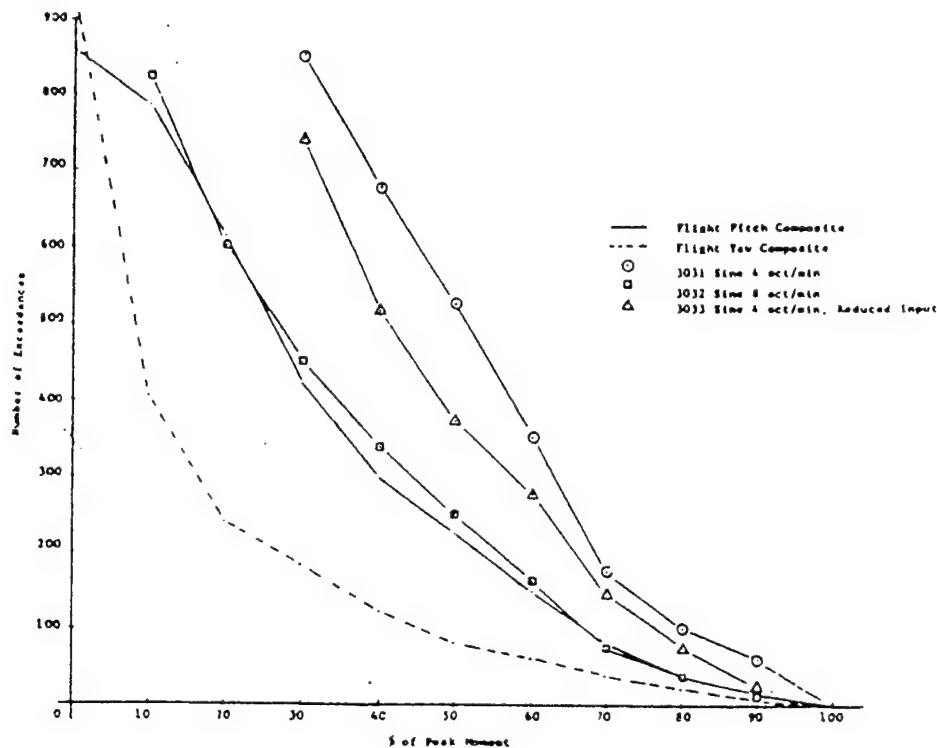
Figure 5.18. Cumulative Amplitude Histogram for Sinusoidal Sweep Tests. From Ref. [18]

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The random dwell histograms shown in Figure 5.19 indicate that the random dwell input resulted in a response with a number of cycles in the low to mid peak moment range that was greater than the flight results. However, this test did not produce the numbers of high level load cycles shown in the flight environment. This could be remedied in the case of the random test by just lengthening the duration of the test. This illustrates one of the benefits of the random approach--the histogram can be shaped to produce the desired number of cycles at the desired load level.

The results of the transient tests, as shown in Figure 5.20, show a significant difference between flight and test. Because the synthesized transient used in this test simulated the liftoff event only, the resulting histograms show fewer cycles at a given load level than the flight histograms, which include all flight events. For an accurate test using the transient method, all flight events must be simulated (such as MECO and BECO). While requiring
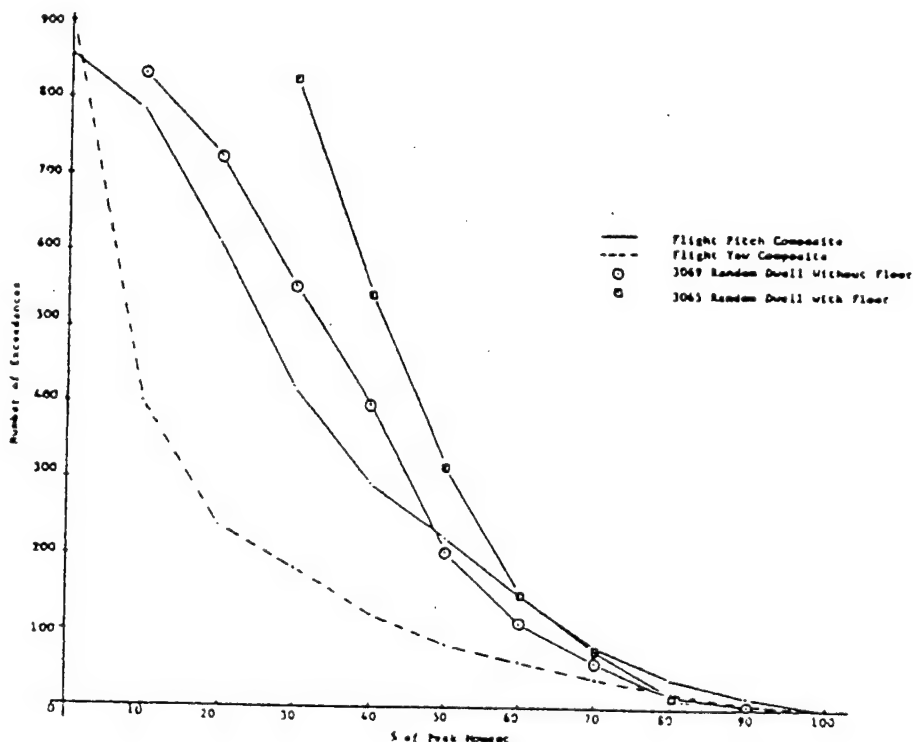
Figure 5.19. Cumulative Amplitude Histogram for Random Dwell Tests. From Ref. [18]

91
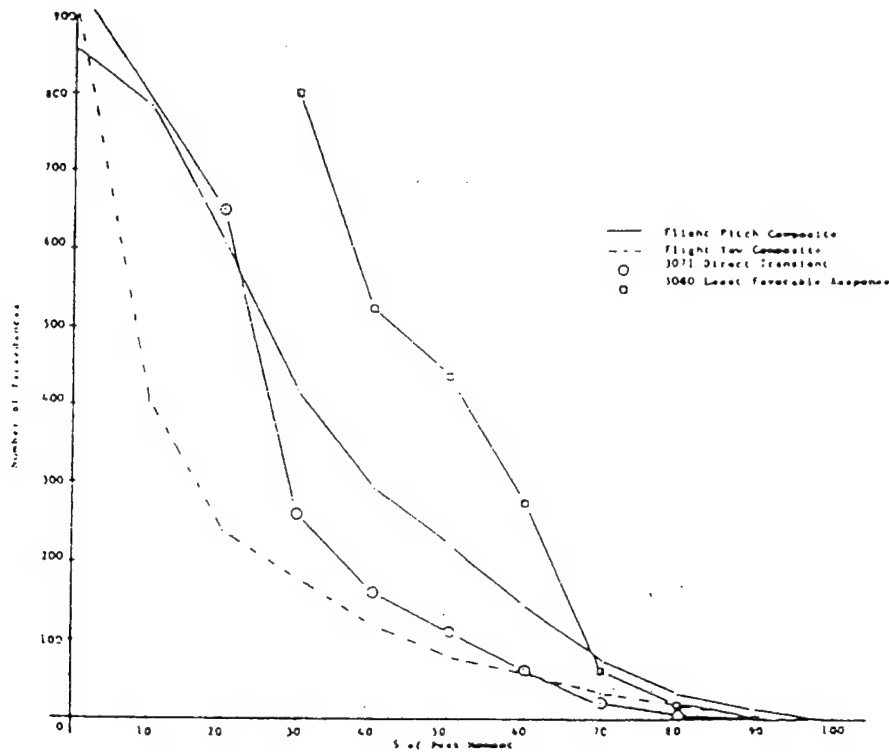
Figure 5.20. Cumulative Amplitude Histogram for Transient Tests. From Ref. [18]

additional transient inputs, which would add to the duration of the required testing, this would not be a significant complication.

As with the random dwell test, the transient tests offer the capability to shape the histogram to more closely approach the flight composite. While the random test histogram can be adjusted by either altering the duration of the test or adjusting the input levels, the transient test histogram can be altered by varying the number of pulses at whatever load levels are desired.

Comparative amplitudes for the various tests are shown in Table 5.8. Results show that the sinusoidal sweep tests adequately tested the primary structure, with response control of other primary structure channels limiting the adapter to only 89% of its peak limit moment. However, the secondary structure was over tested up to 3850% even with peak limiting. The 8 oct/min test reduced the cycle count while maintaining amplitude response of the primary

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Location	Limit Values	Sine 4 oct/min	Sine 8 oct/min	Red Sine 4 oct/min	Random Dwell 1	Random Dwell 2	Direct Transient	Least Fav Res
Bicone Ant Base	17.2	17.0	16.0	17.0	17.2	17.2	12.6	14.2
Rx Antenna Feed	14.0	15.8	13.0	14.0	14.0	7.9	14.0	13.4
Tx Antenna Feed	6.0	6.1	5.8	6.5	5.2	4.8	4.4	2.7
Tx Antenna Attachment	7.1	6.5	6.0	6.0	6.8	4.1	5.4	5.9
Damper Support	6.9	5.6	5.1	5.0	5.6	4.1	4.5	4.82
BAPTA	1.3	3.4	3.4	2.9	1.6	1.5	1.5	0.7
Thrust Cone	0.7	2.0	2.1	1.9	1.1	1.1	1.1	0.6
Tank Mount	0.9	3.1	3.2	2.9	1.1	1.3	1.2	0.5
Receive Reflector	0.4	17.0	16.0	17.0	2.68	7.0	1.4	0.9
Forward Shelf	1.9	2.5	2.5	2.5	3.38	1.8	1.34	1.34
Forward Shelf		2.9	2.8	2.7	1.0	1.3	1.0	1.23
Forward Shelf		8.1	8.0	8.0	2.54	4.3	2.6	3.02
BAPTA Torsion		32K	29K	32K	11.1	10.7	9.0	5.0

Note: All values in g's except torsion in in-lb.

Table 5.8. Comparative Response Amplitudes for Swept Sinusoidal, Random Dwell and Transient Inputs.

structure. The primary structure was adequately tested while over test of the same magnitude occurred on the secondary structure even at the faster 8 oct/min sweep rate. The reduced input level sinusoidal test, which employed a 1/3 g reduction in input in the un-notched regions, resulted in lower response levels on the primary structure as would be expected. The secondary structure was still over tested, indicating that the reduction in input level and the narrowing of the notches did not significantly impact the secondary structural response.

The Table 5.8 data indicates that the random dwell input provided a very good test for the primary structure. The secondary structure was still over tested by a significant margin, in some cases worse than with the swept sinusoidal input. The random dwell input with the floor addition was intended to boost the secondary response in non-resonant regions of the primary structure. Data from the previous test shows, however, that the secondary response is more than adequately excited by the lower input levels.

The transient test resulted in a slight under test of the primary structure. However, the input pulse could be easily modified to increase the response as necessary, as long as this did not result in over test to the secondary structure. In this case, the secondary structure did experience an over test of up to 308% which is considerably less than the 3850% which occurred with the swept sinusoidal input. The least favorable response test resulted in under testing of the primary structure and most of the secondary structure.

(4) Swept Sinusoidal and Modulated Sinusoidal Pulse Comparison. As discussed in Reference 22, loads predictions for the Galileo program showed significant low (below 35 Hz) and mid (35-100 Hz) frequency transient vibration responses during launch events. Because of concerns that slow swept sinusoidal testing would result in excessive resonance buildup and an excessive number of vibration cycles, several alternative approaches were examined by JPL for simulating the low frequency environment. These alternatives excitations included the standard slow swept sinusoidal and a modulated sinusoidal pulse.

In preparation for testing of the Galileo Radioisotope Thermoelectric Generator (RTG), a series of shaker vibration tests were conducted using a model represented by the simplified one and two mass models shown in Figure 5.21. Test comparisons were made of the slow swept sinusoidal and modulated sinusoidal pulse inputs. The modulated sinusoidal pulse was generated as discussed in Chapter IV, Section B.6. The response at resonance and the number of response cycles were determined for the two test cases along with a sinusoidal dwell case for comparison. Because the Galileo load analysis indicated that spacecraft hardware responses did not contain more than five high amplitude cycles for any significant launch vehicle

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event, the modulated sinusoidal pulse used was limited to 5 cycles that exceeded 0.707 times the maximum amplitude cycle.
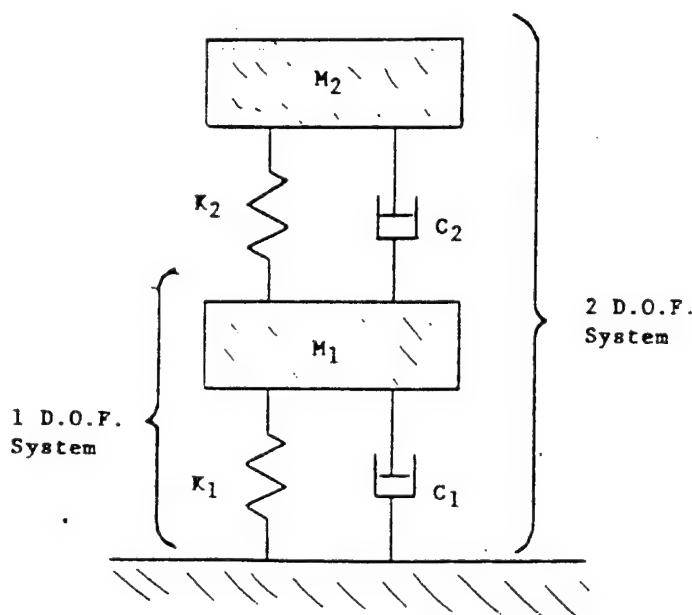
Figure 5.21. Model Used for Galileo RTG Tests. From Ref. [22]

Analytical results for the response of a single degree of freedom system to sinusoidal dwell, sinusoidal sweep and modulated sinusoidal pulse inputs are presented in Tables 5.9 and 5.10. The analytical data shows that the modulated sinusoidal input should result in an appreciably lower response level than the sinusoidal sweep input while the number of cycles is clearly excessive for the sinusoidal sweep input. While the actual test data was not published, according to Reference 22, the analytical and test comparisons showed similar results.

Actual tests were conducted on the RTG and a Magnetometer Boom Assembly using the modulated sinusoidal input. Pulses were space at one-third octave for the RTG test and for the MBA pulses were specified at frequencies for all predicted transients of 2.7 g peak or higher. Again, while actual results were not published, the transient tests were considered successful and were recommended to replace the slow swept sinusoidal tests. However, recent JPL programs

| Test | | System Q | Response G's pk | | |
Method	Condition		$F_n{=}10$ Hz	$F_n{=}25$ Hz	$F_n{=}50$ Hz
Sine Dwell	10 seconds	10	1.0	10	10	10
	30 seconds	10	1.0	10	10	10
Sine Sweep	2 oct/min	10	1.0	9.7	9.5	9.4
	6 oct/min	10	1.0	9.5	9.3	9.0
Modulated Sine Pulses	5 cycles ($\geq$ 0.707 max pulse amplitude)	10	1.0	7.5	7.5	7.5

Table 5.9. Response of a SDOF System to Sinusoidal Dwell, Sinusoidal Sweep and Modulated Sine Pulse Inputs.

| Test | | System Q | Number of Cycles ($\geq$ 0.707 * max response) Resonant Frequency | | |
Method	Condition		10 Hz	25 Hz	50 Hz
Sine Dwell	10 seconds	10	100	250	500
	30 seconds	10	300	750	1500
Sine Sweep	2 oct/min	10	43	108	216
	6 oct/min	10	14	36	72
Modulated Sine Pulses	5 cycles ($\geq$ 0.707 * max pulse amplitude)	10	5	5	5

Table 5.10. Number of Response Cycles for a SDOF System with Sinusoidal Dwell, Sinusoidal Sweep and Modulated Sine Pulse Inputs.

such as Mars Observer and Cassini have not employed the modulated sinusoidal pulse technique. In the case of Cassini, a force limited random vibration test has been adopted.

(5) Shock Spectrum Test Analysis. As discussed in references 8 and 38, Intelsat dynamics tests through the advent of Intelsat V had consisted of 3-axis swept sinusoidal testing and acoustic testing. With Intelsat V, shock spectrum testing was adopted to reduce the risks of over testing during qualification for the Ariane Launch vehicle. The specification for the test was in the form of a shock response spectrum as determined from the peak response of a SDOF oscillator plotted vs. natural frequency. The shape of the SRS depends on the amplitude and frequency content of the input waveform and the damping of the oscillator, as described in Chapter IV, Section A.2. The drive waveform in this case was generated from a 1/6 octave analysis of the synthesized SRS between 10 and 90 Hz. A Q of 10 was assumed in shaping the input SRS.

Shock spectrum results for this test are presented in Table 5.11. While there are no sinusoidal tests to compare to the shock spectrum tests, the response of the shock spectrum test was in the range of the peak expected response. Initially, there was a significant over test ranging from 1.5 to 2.0 times the predicted responses, most likely due to differences in the impedance at the base of the satellite when bolted to a shaker table versus being attached to the launch vehicle adapter. The SRS specification was then lowered to produce responses which matched the predictions of the loads analysis. This lower specification was set at 72% of the original SRS specification. Subsequent data from actual flight measurements on the FM-15 satellite adapter, which is located just above the bolted face to the Ariane adapter, shows that the test SRS is very representative of the actual flight spectra. This flight spectra is shown in Figure 5.22. In this case, the peak excitation of the main axial mode was very near the upper limit for the ground tests and the higher frequency components are all below the test specification limits. This indicates that the shock spectrum method is a realistic simulation of the flight loads.

LOCATION	PREDICTED RESPONSE (g)	MEASURED RESPONSE (g)
4 GHz Reflector	7-9	15
6 GHz Reflector	4.2	8
West Spot Reflector	4.3	11
Maritime Antenna	4.6	12
Momentum Wheel	2.4	4
Thruster Clusters	14.5-16	32
Propellant Tanks	7.2-9	9-10
Comm Transponder (N Side)	8	8
Comm Transponder (S Side)	8-9.5	9

Table 5.11. Intelsat V Shock Spectrum Response Data.

(6) Fast Swept Sinusoidal Test Analysis. The execution of the swept sinusoidal test with a significantly increased sweep rate is discussed in references 7 and 39. This approach is offered as an option when a higher cost digital controller is not available for synthesized time waveform simulation of the transient environment. The intent is to reduce the potential of over testing inherent with the slowly swept sinusoidal input by increasing the sweep rate. The goal is to excite the test article at a higher sweep rate and adjust the input level to provide the desired response. The sweep rate is determined by the number of cycles desired as determined from flight acceleration time histories.

While an example of this approach was discussed, no experimental data was provided to support the conclusions of the paper. This approach would clearly result in a reduction of the number of cycles to which the test article is exposed. Based on the relationships presented in Chapter IV, Section C.2 and as verified by the moderate increase in sweep rate discussed in
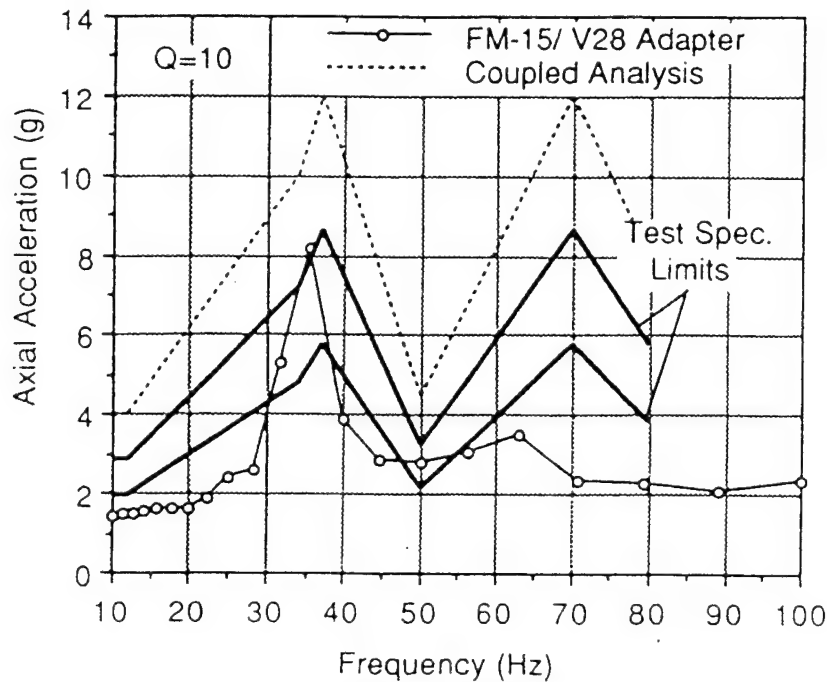
Figure 5.22. Flight Shock Response Spectrum from Intelsat V
Satellite Adapter. From Ref. [8].

study number (3), this test method has some merit. However, with the current availability of
digital controllers, there has been little application of the fast swept sinusoidal method.

(7) Multi-Axis Vibration System Applications. Each of the previous
dynamics qualification studies and tests were based on the application of the vibration input in a
single axis. While testing may have been conducted in all three axis, independently, vibration
systems were typically not capable of excitation in multiple axes simultaneously. The relatively
recent development of multi-axis vibration systems (MAVIS) has enabled the excitation of any
rigid body motion in three axis with 6 degrees of freedom. MAVIS employment has taken place
almost exclusively in Europe, where multi-axis vibration test simulators previously used for
earthquake simulation have been employed to simulate launch vehicle multi-axis transients. The
multi-axis testing is considered to provide a more realistic simulation of the actual environment than
the classical single axis approach by providing correct super positioning of the multi-axial
structural loads.

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Several studies of the potential application of MAVIS to spacecraft qualification testing have been performed. One such study, presented in Reference 40, discussed the execution of transient testing in 6 degrees of freedom on a MAVIS table using a structural model of the DFS-KOPERNIKUS telecommunications satellite. Test results comparing a transient excitation of the DFS mass dummy at 50% amplitude with the desired specification levels are shown in Figure 5.23. The plots show that the multi-axis system was able to very closely simulate the environment in all 6 degrees of freedom. Slight deviations in the test results were due to control system issues which required some modification to the control network.

While the feasibility of multi-axis transient testing has been shown, whether such testing provides significant advantages to the qualification of a space vehicle is yet to be proven. With the current focus on less expensive test methods, the cost of a multi-axis system may not be justifiable to most spacecraft manufacturers.
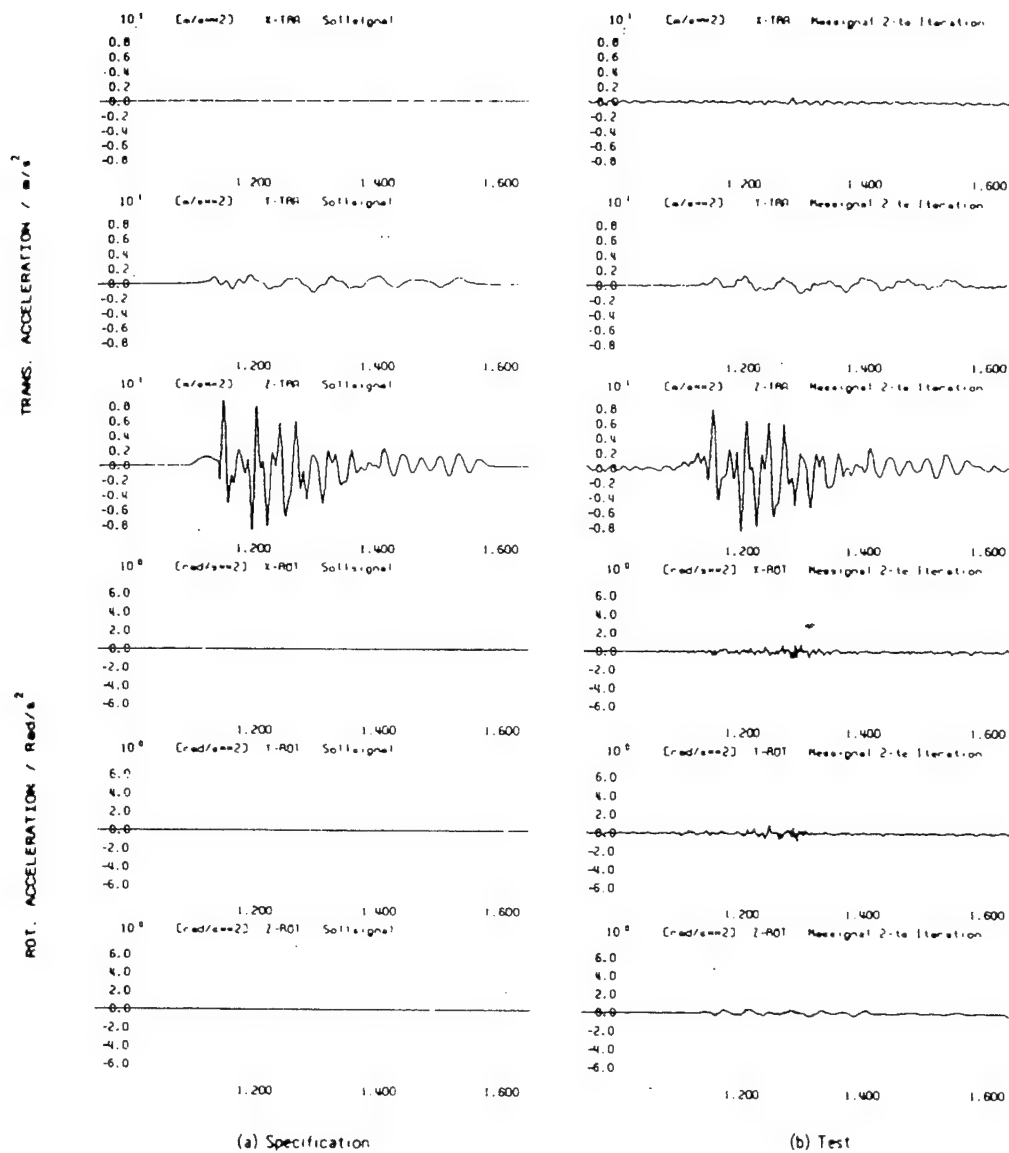
Figure 5.23. MAVIS Test Transient Response Data. From Ref. [40].

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VI.   CONCLUSIONS AND RECOMMENDATIONS

A.   CONCLUSIONS

1.   Acceptance Testing

Test failure and on-orbit performance data available for this study indicate that acoustic testing at the vehicle level provides an adequate workmanship and materials defect screen, especially for mature programs. The data does not indicate that additional dynamics tests such as sinusoidal or random vibration result in either an increase in the detection of failures during test or a reduction in the early flight failure rate. Data from other test effectiveness studies supports this conclusion. Additional data is required to generate a more thorough understanding of acceptance test effectiveness as well as to provide statistical significance.

The experimental studies analyzed for this report provide no information on the effectiveness of acoustic testing relative to sinusoidal and random vibration testing. However, the JPL study clearly indicates that random vibration at appropriate levels is far more perceptive than swept sinusoidal testing in exposing workmanship defects. As indicated by the Topaz II test data, this result is demonstrated despite the fact that the random vibration test results in far lower response levels throughout a typical space structure. Consequently, the random test at the vehicle level provides greater test effectiveness with lower risk of damage to hardware. While there is no experimental data to indicate how acoustic testing would rate, the failure data as discussed above indicates that a similar acoustic experiment should result in ever greater failure detection rates.

2.   Qualification Testing

The studies and tests surveyed for the analysis of vehicle level qualification testing clearly indicate that the single axis slow swept sinusoidal test provides an unrealistic simulation of the flight environment. Even when notching is applied, the response levels especially for secondary structure, can exceed actual flight values by significant margins. At the same time, other locations can be significantly under tested. In addition, the slow swept sinusoidal test exposes hardware to significantly greater numbers of cycles than the flight environment.

The studies surveyed almost uniformly indicate that a transient input developed either directly from flight data or synthesized from the shock response spectrum very accurately simulates the flight environment. The transient approach to dynamic qualification testing minimizes the risk of over test while producing realistic responses with regard to magnitude and numbers of cycles. With current digital control systems, such testing is relatively easy to conduct. Transient testing clearly demonstrates significant utility when applied to spacecraft qualification testing and to the validation of analytical models for complete design verification.

While offering simplicity in execution as well as a reduction in the potential for over testing, there is not enough information to validate the effectiveness of the fast swept sinusoidal approach for qualification testing. Similarly, MAVIS testing, while offering a more accurate simulation of the launch environment, has not yet been developed to the point that significant improvements in effectiveness can be shown.

B.   RECOMMENDATIONS

The following recommendations are provided as a result of this study:

1.   The Space Systems Engineering Database (SSED) developed by the Aerospace Corporation under US Air Force sponsorship is a useful resource for the analysis of spacecraft testing as well as a variety of other issues. The information currently available in the SSED is an excellent start but should be supplemented with additional data, especially from non-DOD programs which have traditionally employed different test approaches. As the commercial spacecraft manufacturing business is highly competitive, it is understandable that data from commercial programs is not available. However, data from government programs should be available and would greatly benefit the entire US space industry. In particular, it is recommended that NASA and JPL data be added to the SSED as soon as possible.

2.   The utility of test data such as that in the SSED would be greatly enhanced if test programs systematically documented failures with greater detail. Specifically, it is

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recommended that data relating failures to the specific test in which they occurred along with test levels and durations be recorded whenever possible for DOD and other government sponsored test programs. Such data would greatly assist in the analysis of test effectiveness. While the collection of this type of data would increase costs in a test program, the long term benefits are significant.

3. An acoustic workmanship study such as that conducted by JPL for sinusoidal and random vibration inputs is recommended to further analyze the effectiveness of acoustic tests for spacecraft acceptance.

4. A more detailed study regarding fast swept sinusoidal testing for vehicle level qualification tests is recommended to determine if such an approach would provide an additional alternative to the slow swept sinusoidal test for spacecraft qualification.

5. With the continued development of multi-axis vibration systems, it is recommended that additional data is collected to determine if the effectiveness of spacecraft qualification testing is increased by the use of multi-axis techniques.

6. The MIL-STD-1540C focus on acoustic testing at the vehicle level appears to be validated. However, the MIL-STD provides little discussion of the need for space vehicle qualification in the low frequency quasi-static environment. As this environment has been shown to be a significant design driver for both primary and secondary structure, it is recommended that MIL-STD-1540C incorporate more detail with regard to qualification of the low frequency environment, either through analysis or test.

APPENDIX A. TYPICAL LAUNCH ENVIRONMENT
SPECIFICATIONS

A.   Delta 3000 Series Launch Vehicles.

Thrust Axis	Lateral Axis
+2.8/ -1.4	3.4 for Payloads < 800 lb 2.9 for P/L 800-1200 lb 2.4 for P/L > 1200 lb

Table A.1. Delta 3000 Series Liftoff Load Factors. From Ref. [14].

Vehicle Configuration	Axial (G)	Lateral (G)	Payload Wt. (lb)
3913	7.8 +/- 4.6	+/- 1.0	> 1100
3920	7.0 +/- 4.1	+/- 1.0	> 4000
3914	7.5 +/- 4.6	+/- 1.0	> 2000
3910	8.1 +/- 4.1	+/- 1.0	>2500

Table A.2. Delta 3000 Series Limit Load Factors at MECO
at Spacecraft Center of Gravity. From Ref. [14].

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100 100-800 800-4000	15 +8.7 dB/oct 200	11 +8.7 dB/oct 140

Table A.3. Delta 3000 Series Shock Response Spectra, Q=10. From Ref. [14].

Frequency (Hz)	ASD Level (G^2/Hz)
20	.0016
20-300	+4 dB/oct
300-700	.06
700-2000	-3 dB/oct
2000	.021
Overall	8.7 G_{rms}

Table A.4. Delta 3000 Series Spacecraft Random Vibration Limit Levels. From Ref. [14].

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	122	119
32	123	120
40	125	122
50	126	123
63	127	124
80	127	124
100	130	127
125	132	129
160	134	131
200	134	131
250	135	132
315	137	134
400	137	134
500	142	139
630	137	134
800	134	131
1000	132	129
1250	130	127
1600	130	127
2000	129	126
2500	127	124
3150	127	124
4000	125	122
5000	125	122
6300	123	120
8000	121	118
10000	121	118
Overall	147	144

Table A.5. Delta 3000 Series Acoustic Test Levels
(Inside Payload Fairing). From Ref. [14].

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B.   Shuttle Transportation System (STS).

One Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Protoflight	Acceptance
25	122.0	119.0
32	125.0	122.0
40	128.0	125.0
50	130.5	127.5
63	131.5	128.5
80	132.0	129.0
100	132.0	129.0
125	132.0	129.0
160	131.5	128.5
200	130.5	127.5
250	130.0	127.0
315	129.0	126.0
400	128.0	125.0
500	127.0	124.0
630	126.0	123.0
800	124.5	121.5
1000	123.0	120.0
1250	121.5	118.5
1600	119.5	116.5
2000	118.5	115.5
2500	116.0	113.0
3150	114.5	111.5
4000	112.5	109.5
5000	111.0	108.0
6300	109.0	106.0
8000	107.5	104.5
10000	106.0	103.0
Overall	142	139

Table A.6. STS Cargo Bay Acoustic Test Levels
(Up to 9 ft diameter payloads). From Ref. [14].

C.   Atlas I, II and IIA Launch Vehicles.

Event	Axial	Lateral
Launch (Atlas I, II,IIA)	1.2 +/- 1.5	+/- 1.0
Launch (Atlas IIAS)	1.3 +/- 1.8	+/- 2.0
Flight Winds	2.2 +/- 0.3	0.4 +/- 1.2
BECO (max axial)	5.5 +/- 0.5	+/- 0.5
BECO/BPJ (max lateral)	2.5-1.0 +/- 1.0	+/- 2.0
SECO	2.0-0.0 +/- 0.4	+/- 0.3
MECO	4.0-0.0 +/- 0.5 0.0 +/- 2.0	+/- 0.5

BECO = Booster Engine Cut-off
BPJ = Booster Package Jettison
SECO = Sustainer Engine Cut-off
MECO = Main Engine Cut-off

Table A.7. Atlas I, II, IIA and IIAS Limit Load Factors
at Spacecraft Center of Gravity. From Ref. [14].

Frequency (Hz)	ASD Level (G^2/Hz)
20	.00048
20-200	+9 dB/oct
200-500	.03
200-2000	-4.5 dB/oct
2000	.0038
Overall	5.3 G_{rms}

Table A.8. Atlas I, II, IIA and IIAS Spacecraft Random
Vibration Limit Levels. From Ref. [14].

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Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	28	20
100-1600	+10 dB/oct	+10 dB/oct
1600-4000	2800	2000

Table A.9. Atlas I, II, IIA and IIAS Spacecraft Separation Shock Response Spectrum, Q=10. From Ref. [14].

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	20	14
100-500	+5.4 dB/oct	+5.4 dB/oct
500-4000	84	60

Table A.10. Atlas I, II, IIA and IIAS Payload Fairing and Insulation Jettison Shock Response Spectrum, Q=10. From Ref. [14].

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa			
	Without Acoustic Blankets		With Acoustic Blankets	
	Qualification	Acceptance	Qualification	Acceptance
25	121	118	121	118
32	123	120	123	120
40	124.5	121.5	124.5	121.5
50	126	123	126	123
63	128	125	128	125
80	129	126	129	126
100	130.5	127.5	130	127
125	132	129	131	128
160	132.5	129.5	131	128
200	133.5	130.5	131.5	128.5
250	134	131	131	128
315	133	130	129	126
400	132	129	127	124
500	131	128	125	122
630	129.5	126.5	123.5	120.5
800	127	124	121	118
1000	125	122	119	116
1250	122	119	116	113
1600	120	117	114	111
2000	119	116	113	110
2500	118.5	115.5	112.5	109.5
3150	118	115	112	109
4000	117.5	114.5	111.5	108.5
5000	117	114	111	108
6300	116.5	113.5	110.5	107.5
8000	116	113	110	107
10000	115.5	112.5	109.5	106.5
Overall	143	140	140	137

Table A.11. Atlas I, II, and IIA Acoustic Test Levels (Inside 11 ft. Diameter Payload Fairing). From Ref. [14].

LIST OF REFERENCES

1. MIL-STD-1540C, *Test Requirements for Launch, Upper-Stage, and Space Vehicles,* 15 September 1994.

2. Hamberg, O., Tosney, W., "The Effectiveness of Satellite Environmental Acceptance Tests," *The Journal of Environmental Sciences,* 1988.

3. Tosney, W., "Spacecraft Test Effectiveness Brief," The Aerospace Corporation, 1993.

4. "GOES Spacecraft Testing Process Brief," NASA Goddard Space Flight Center.

5. Harris, C.M. and Crede, C.E., *Shock and Vibration Handbook,* McGraw-Hill, Inc., 1976.

6. Moening, C.J., "An Overview of Shock, Vibration and Acoustic Environments," The Aerospace Corporation, 1988.

7. Xian, H.Y., "A Simulated Method of Transient Environment--The Fast Swept Sine Method," *Proceedings of the Institute of Environmental Sciences,* 1981.

8. Lomas, N.S., Schrantz, P.R., "An Evaluation of Low Frequency Shock Spectrum Testing For Intelsat V / Ariane Compatibility," *International Symposium, Environmental Testing For Space Programs, Test Facilities and Methods,* 26-29, June 1990.

9. Hershfeld, D.J., "Avoiding Unrealistic Overtest By Notching the Vibration Test Specification," *Proceedings of the Institute of Environmental Sciences,* 1974.

10. Herzberg, R.J.,"Structural Dynamic Testing Considerations for Large Space Vehicles," Lockheed Missiles and Space Company, March 1974.

11. Ory, H., Hornung, E., "Real Launch Vehicle Flight Loads and Their Simulation For Spacecraft Development," ERNO Raumfahrttechnik GmbH, Bremen, Germany.

12. Slone, R.M., "System Level Acoustic Testing," *Proceedings of the Institute of Environmental Sciences.*

13. MIL-HDBK-340, *Application Guidelines for MIL-STD-1540C; Test Requirements for Launch, Upper Stage, and Space Vehicles.* June 1995.

14. GEVS-SE, *General Environmental Verification Specification for STS and ELV Payloads, Subsystems, and Components,* NASA Goddard Space Flight Center, January 1990.

15. Coladonato, R.J., "Derivation of the Sinusoidal Sweep Vibration Test Levels for the Cosmic Background Explorer," *Proceedings of the 13th Aerospace Testing Seminar,* 1991.

16. "Atlas II DOD User's Mission Planning Guide," General Dynamics Space Systems Division, December 1988.

17. Voorhees, C., McKee, L., "Dynamic Testing of Spacecraft Structures," *Sound and Vibration ,* Vol. 24, Issue 1, January 1990.

18. Fowler, J.R., Zuziak, R.J., DesForges, D.T. and Schrantz, P.R., "Alternatives to Notched Sine Testing for Spacecraft and Other Large Systems," *Proceedings of the Institute of Environmental Sciences*, April 1977.

19. Kern, D., Barrett, P., "Rationale for Cassini Spacecraft Random Vibration Test," Jet Propulsion Laboratory Memorandum, 20 February 1996.

20. Hine, M.J., "Swept Sine Vibration Test Conservatism," *Journal of the Institute of Environmental Sciences*, November-December 1995.

21. Agrawal, B.N., "Stability of an Automatic Notch Control System in Spacecraft Testing," *Shock and Vibration Bulletin*, January 1972.

22. Kern, D., "Vibration Problems-Transient Vibration Test Criteria," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California.

23. Schlue, J., "Flight Payloads Environmental Approach," *Proceedings of the Institute of Environmental Sciences*.

24. "Sine Vibration Testing Brief," Dynamics Environments Group, Jet Propulsion Laboratory, June 1993.

25. Boatman, D., Scharton, T., Hershfeld, D. and Larkin, P., "Vibration and Acoustic Testing of Topex/Poseidon," *Journal of the 17th Space Simulation Conference*, 9-12 November 1992.

26. Milder, G., "Viking 1975 Orbiter DTM/Lander Dynamic Test Model Dynamic Environmental Testing: An Overview," *Proceedings of the Institute of Environmental Sciences, Vol 2*, 14-16 April 1975.

27. Pike, H.S., Day, R., Gatta, J., "AUSSAT-B Spacecraft Qualification and Acceptance Testing." *Proceedings of the 13th Aerospace Testing Seminar*, 8-10 October, 1991.

28. Ginsdorf, T., "Tailoring and Test Effectiveness," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California.

29. Lang, P., Card, M.,"Application of Test Effectiveness in Spacecraft Testing," *Proceedings of the Annual Reliability and Maintainability Symposium*, 1995.

30. Ackerman, A. "Refining Test Effectiveness Through Systems And Cost-Of-Quality Analysis," *Proceedings of the Institute of Environmental Sciences*.

31. Smith, D.A., "Evaluation of Spacecraft Acceptance Test Effectiveness," *Proceedings of the Institute of Environmental Sciences*.

32. Smith, D.A., "System Level Acoustic Test Effectiveness," *Proceedings of the Institute of Environmental Sciences*, 1986.

33. Hine, M.J., "Comparison of Workmanship Vibration Test Methods," Report D-12903, Jet Propulsion Laboratory, 1995.

34. Raney, Elisa Anne, *Vibration Analysis of Topaz II*, Master's Thesis, Naval Postgraduate School, June 1995.

35. Campbell, Sheryl Elaine, *Random Vibration Analysis of the Topaz II Nuclear Reactor Power System,* Master's Thesis, Naval Postgraduate School, June 1995.

36. "Design Verification of Large Spacecraft." MBB Erno Report, Intelsat Contract No. Intel 276, MBB/ERNO Raumfahrttechnik GMBH, Bremen, Germany, July 1985.

37. Hine, M.J., "Evaluation of Conservatism in Low Frequency VibrationTest Control," Jet Propulsion Laboratory, Pasadena, California.

38. Lomas, N. and Mullin, J., "Low-Frequency Shock Spectrum Testing of Intelsat V Spacecraft,*" Proceedings of the Institute of Environmental Sciences*, March 1990.

39. Clevenson, S., Martin, D., Pearson, J., "Representation of Transient Sinusoids in the Environmental Vibration Tests for Spacecraft," *Proceedings of the Institute of Environmental Sciences*, 1965.

40. Muhlbauer, K., Schildt, U., Raasch, W. "Transient Vibration Testing of Spacecraft Structures, Experience With Multi-Axial Base Motion and Combined Base Motion and Point Force Excitation,*" Proceedings of the International Conference on Structures and Mechanical Testing*, 19-21 October 1988.

41. Melancon, P.S., "Fleet Satellite Communications Program." 1980.

BIBLIOGRAPHY

Ankrom, M. and Vicari, L., "Automated Process for Notched Sine Testing," *Proceedings of the 14th Aerospace Testing Seminar*, 1993.

Barrett, S., Maytum, B., "The Development of Sine Vibration Test Requirements For Viking Lander Capsule Components," *Proceedings of the Institute of Environmental Sciences*, 28 April-1 May 1974.

Hoffman, A.R., "Galileo Environmental Test And Analysis Program Summary." *Proceedings of the Institute of Environmental Sciences*, 8-10 October 1991.

Hoffman, A.R., Schlue, J.W., "Galileo Spacecraft System Level Environmental Test Results." *Proceedings of the 9th Aerospace Testing Seminar*, 15-17 October 1985.

Kern, D., "Mars Observer Sine Vibration Testing Brief," Jet Propulsion Laboratory, California Institute of Technology, 9 February 1988.

Lyon, R.H., *Random Noise and Vibration In Space Vehicles,* The Shock and Vibration Information Center, United States Department of Defense, 1967.

O'Connell, M.R., "Effectiveness of Spacecraft Sine Vibration Testing Test Criteria," Report TETA TO-0028. Jet Propulsion Laboratory, 1994.

Swales, T.G., Alexander, J. and Luzier, R., "ATS-F Notching Criteria as Determined by Analysis and Test," *Proceedings of the Institute of Environmental Sciences*, 28 April - 1 May 1974.

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